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MAR 79 J B WATSON, D A GLAESER, F L HARVEY N62269-77-C-0366
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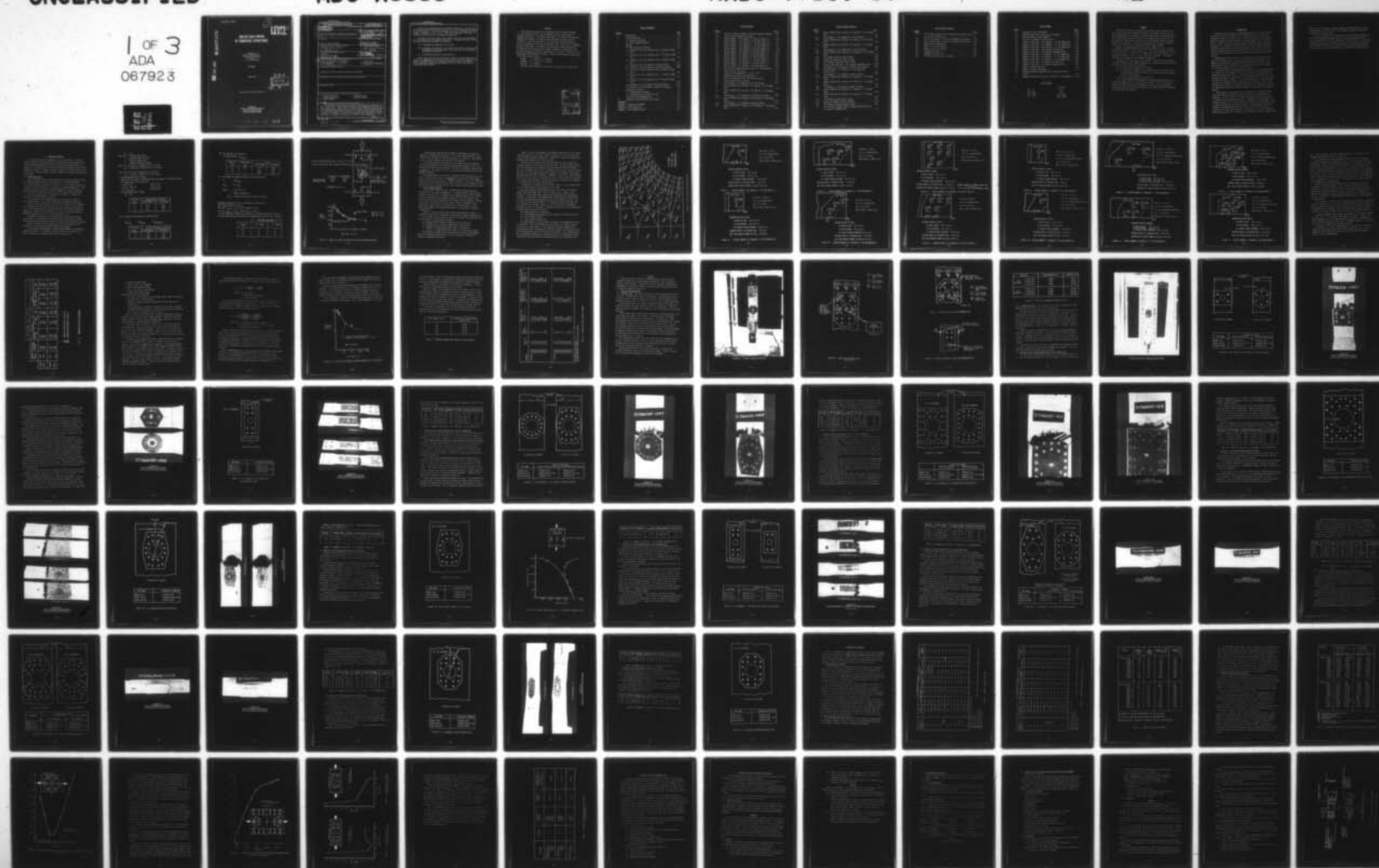

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BOLTED FIELD REPAIR OF COMPOSITE STRUCTURES

J. C. Watson, et al.
McDonnell Douglas Corporation
McDonnell Aircraft Company
P.O. Box 516
St. Louis, Missouri 63166

Final Report

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Mechanically fastened, field repairs for graphite/epoxy laminates 3/16 and 1/2 inch thick with through-the-thickness hole damage have been successfully demonstrated. These repairs (titanium alloy patches and backing plates) were developed for application on fuel cell composite wing surfaces and can be installed by maintenance personnel in the field using available equipment, materials, and methods. A total of 28 specimens were fabricated and tested statically to evaluate effectiveness of selected repair designs.

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Initially, one static tension test conducted on each repair established the better design for each damage configuration. Subsequent testing, utilizing the selected repair design, was conducted:

(a) To demonstrate repeatability of results.

(b) To evaluate the strength of the repair when there is as much as 20° misalignment between the repair patch centerline and the laminate principal load direction.

(c) To determine compression characteristics.

Static tests indicated that mechanically fastened (bolted) patches can restore the damaged skin to a gross failure strain in excess of 4,000 μin./in. Results also demonstrated that composite skin repairs can be performed by field maintenance personnel using familiar tools, as easily and quickly as current metal skin repairs.

FOREWORD

This report was prepared by the McDonnell Aircraft Company (MCAIR), St. Louis, Missouri, under Contract N62269-77-C-0366, "Field Repair of Composite Structures". The design, development, and testing of mechanically fastened field repairs for graphite/epoxy laminates in wing structures is reported. The work described in this report was conducted during the period 27 September 1977 through 30 September 1978. The report is catalogued by MCAIR as MDC A5583. The contract was administered under the direction of the Naval Air Development Center (NADC), Warminster, Pennsylvania, with Mr. Mark Libeskind as the NADC Program Monitor. Mr. James C. Watson was the Engineering Program Manager for MCAIR. Principal MCAIR technical contributors to the work reported include:

Design: J. B. Watson, D. A. Glaeser

Strength: F. L. Harvey, W. T. Fujimoto

Testing: V. E. Padilla

This report was submitted by the author for publication in March 1979.

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SUMMARY

Mechanically fastened, field repairs for graphite/epoxy laminates 3/16 and 1/2 inch thick with through-the-thickness hole damage have been successfully demonstrated. These repairs (titanium alloy patches and backing plates) were developed for application on fuel cell composite wing surfaces and can be installed by maintenance personnel in the field using available equipment, materials, and methods. A total of 28 specimens were fabricated and tested statically to evaluate effectiveness of selected repair designs.

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1. INTRODUCTION

Structural repairs applied to damaged graphite/epoxy structures usually have been in the form of graphite/epoxy reinforcements either cocured or secondarily bonded over damaged areas. Materials utilized in these repairs require refrigerated storage and heat and vacuum equipment for processing. Also, to insure structural reliability, bonded composite repairs require application in a relatively clean environment. Field-type composite repairs performed without benefit of autoclave pressures and heat sources must not only be configured for structural effectiveness, but must take into account utilization of available tools and personnel probably unfamiliar with composites.

Emerging Navy fleet aircraft are using graphite/epoxy structures which consist of relatively thick composite skins bolted to substructure. The laminates are designed for high strength and operate at low strain levels. These features allow consideration of mechanically fastened metal patches, which are particularly suited to field level repair.

An earlier MCAIR investigation demonstrated that significant improvement in strength levels of damaged skin panels can be achieved with mechanically fastened titanium repair plates. The investigation, intended primarily to evaluate the effects of edge distance in a hole-damaged fuel cell, also served to develop a sealing technique. The strength attained by the repaired panel specimen was 57% greater than the unrepaired panel strength. Thus, it was believed that mechanically fastened repair plates can be designed to replace the damaged material and re-establish strength, in addition to providing fuel sealing.

Specifically, the objective of this program was to develop and demonstrate mechanically fastened field repair techniques for graphite/epoxy laminates. The work consisted of design, fabrication, and testing efforts, in conjunction with supporting analyses. Through-the-thickness damage consisting of holes 1.0, 2.5, and 4.0 inches in diameter were considered. Since a design allowable strain of 4,000 μin./in. at design ultimate load was the baseline for undamaged laminates, recovery of an equivalent strain was a primary goal for the repair concepts. Misalignment of the repair patch by maintenance personnel not familiar with the reference systems or primary load

paths was also considered. Finally, repairs and repeatability of results were demonstrated on 3/16 and 1/2 inch thickness specimens representative of AV-8B and F-18 wing covers.

This final technical report describes the work performed during the 12-month period ending 30 September 1978 under Contract N62269-77-C-0366. A detailed discussion of all work accomplished under the contract is presented including repair designs, analyses, test data, in addition to conclusions and recommendations related to program results and their implications for future work in the area of field repairs of graphite/epoxy wing structures.

2. TECHNICAL APPROACH

The repair concepts were designed in a two-step process. The first step was based on a coarse approach of determining how many fasteners were required and configuring repair plates to contain the required number of fasteners. The second step was to build finite element models of the initial designs and adjust the original designs to improve the strain patterns in the repair area. Design adjustments included changes in the number and relocation of fasteners.

2.1 DESIGN & ANALYSIS

Since the patches must be easily applied in the field, composite patches and bonded attachments were not considered for the following reasons: (1) Precured composite patches are difficult to fit to a wing contour, and a composite patch could itself be susceptable to handling damage. (2) Bonding must be done in a relatively clean environment to insure a quality repair, and this is difficult to attain on a fuel-soaked wing in the field.

Therefore, metal patches and fasteners were chosen. Titanium was selected to preclude galvanic corrosion.

The outer patch was a standard gage, and thick enough to accept a flush head fastener. The edges were beveled for aerodynamic smoothness.

Internal backing plates were used to hold nut plates for the fasteners, and when mecessary these backing plates were thick enough to carry load as well. All backing plates were inserted from the outer surface through the damage hole, within the depth confines of the wing. The AV-8B wing has a minimum working depth of 6 in. The F-18 wing has 2.5 in.

In addition to restoring structural integrity, the repairs provided fuel sealing. To accomplish this, fasteners with O-ring seals were used. The patch-to-wing-skin faying surfaces can be coated with sealant at assembly.

Screws with plate nuts were selected so that, in the event a leak should occur, the patch can be easily removed and resealed.

For the initial determination of fastener requirements, the following approach was taken: (1) Determine the load carried in the parent skin (panel laminate) at the design strain level of 4000 μin./in.

$N_x = \varepsilon E t$

where N_x = running load (lb/in.)

ε = material strain (in./in.)

E = Young's modulus (psi)

t = Material thickness (inches)

Since E for the 3/16 in. laminate is 9.05×10^6 psi:

$N_x = (.004)(9.05 \times 10^6)(.1872) = 6770$ lb/in.

Since E for the 1/2 in. laminate is 10×10^6 psi:

$N_x = (.004)(10 \times 10^6)(.4784) = 19150$ lb/in.

- For unrepaired failure strains:

  Based on YAV-8B tests of unrepaired flaws (Ref Figure 2-1) failure strains for damage sizes to be tested are:

1 in. dia. hole	3200 μin./in.
2-1/2 in. dia. hole	2100 μin./in.
4 in. dia. hole	1750 μin./in.

- For unrepaired failure loads:

  $N_x = \varepsilon E t$

HOLE SIZE (INCH)	FAILURE LOAD, N_x (LB/IN)	
	3/16" PANEL	1/2" PANEL
1.0	5420	15300
2.5	3550	10050
4.0	2960	8370

(2) For splice loads required in repair patches:

$N_{x_{splice}} = N_{x_{parent}} - N_{x_{unrepaired}}$

HOLE SIZE (INCH)	SPLICE LOAD $N_{x_{splice}}$ (LB/IN)	
	3/16" PANEL	1/2" PANEL
1.0	1350	3850
2.5	3220	9100
4.0	3810	10780

(3) For the load to be spliced:

$P = (\text{Panel width})\ (N_{x_{splice}})$

HOLE SIZE (IN.)	PANEL WIDTH (IN.)	P (LB)	
		3/16" PANEL	1/2" PANEL
1.0	3-1/2	4725	13480
2.5	5	16100	45500
4.0	8	30480	86240

(4) For number of fasteners required in the repair patch:

$N = 2P/P_{bolt}$

$P_{bolt} = d\ t\ F_{br_u}$

Where d = bolt dia, in.

t = panel thickness, in.

F_{br_u} = ultimate panel bearing stress, 50,000 psi

Fasteners selected are 1/4 in. sealing head titanium screw.

Bolt shear strength is 4660 lb.

For 3/16", P_{bolt} = (.25) (.1872)(50,000) = 2340 lb

For 1/2", P_{bolt} = (.25) (.4784)(50,000) = 5980 lb

(5) The first-pass estimate of the number of fasteners required for each of the hole sizes and panel thicknesses is summarized below for 1/4 in. fasteners:

	FASTENERS REQUIRED, N	
HOLE SIZE (in.)	3/16" PANEL	1/2" PANEL
1.0	6	6
2.5	14	16
4.0	28	30

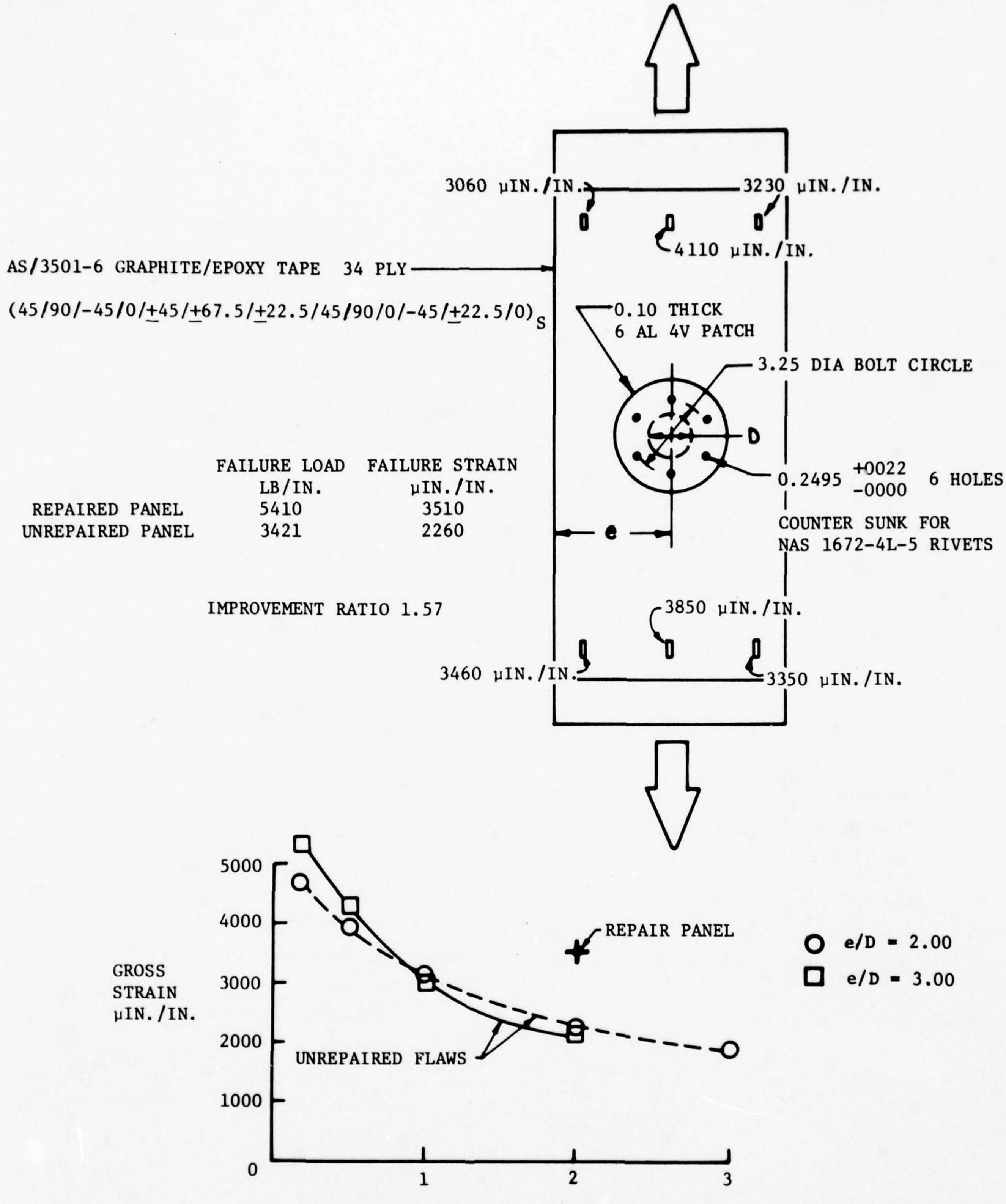


FIGURE 2-1 EFFECTS OF FLAWS AND REPAIR ON WING SKIN LAMINATE SPECIMEN

Initial design concepts were developed to incorporate the required numbers of fasteners indicated above. For a second design approach, other concepts were developed for repairs using 5/16 in. fasteners. This allowed the use of fewer fasteners and different bolt patterns.

The titanium patches were made from the thinnest stock size material which could accomodate the countersunk sealing head fasteners. For 1/4 in. fasteners a 0.140 in. patch was used. The 5/16 in. fasteners required 0.160 in. material. All fastener holes in the patches were defined with close fit $^{+0.003}_{-0.000}$ tolerances.

The backing plates for the 3/16 in. laminates were not required to carry load. Therefore thin (0.040) titanium material was used and the fastener holes were defined with loose fit $^{+0.006}_{-0.000}$ tolerances. The 1/2 in. laminates required structural backing plates to prevent excessive bolt bending. For these repairs 0.140 in. backing plates and close tolerance fastener holes were used.

2.2 FINITE ELEMENT MODELING

In the detailed analysis stage, NASTRAN finite element models were developed for each repair configuration. The NASTRAN finite element models utilized flat anisotropic rectangular and triangular elements for the panels, isotropic rectangular and triangular elements for the repair patches and back-up plates, and one-dimensional beam elements for the fasteners. Bending of the panel and patches was ignored, and the repair configuration was idealized as planar "sheets" interconnected by flexible beams. In this manner, the complex three-dimensional behavior of the joint was reduced to two-dimensions.

Figure 2-2 shows a typical finite element idealization of a panel. A relatively coarse finite element mesh was utilized to minimize modeling costs.

The strength criteria initially used for the repairs were taken from the YAV-8B composite wing development program:

(a) Maximum average bearing in composite skin = 50,000 psi

(b) Maximum strain at edge of damage hole = 10,000 μin./in.

(c) Maximum bolt loads within rated limits of bolt used.

(1/4 in. = 12,800 double shear; 5/16 in. = 19,300 double shear)

The preliminary design concepts were analyzed in detail using NASTRAN and, when indicated, modifications in either the number of fasteners or geometric arrangements were made.

Figures 2-3 thru 2-14 summarize the NASTRAN predictions for the internal loads and strains for the repair concepts selected for the various hole diameter and panel thickness combinations. Appendix A contains the engineering drawings which describe the patches in detail.

The strain at the edge of the hole was assumed to be the same as the centroidal strain of the element at the hole edge. The bolt stiffness values used for the analyses were obtained from the empirical formula of Voight, Ref. (2). Fasteners in the 3/16 in. laminates with 0.140 or 0.160 in. patches and 0.040 in. thick back-up plates were assumed to be loaded in single shear (back-up plate holes were oversized). Fasteners in the 1/2 in. thick laminates with the 0.140 or 0.160 in. patches and 0.140 in. back-up plates were assumed to be loaded in double shear. Fastener clearance was assumed to be negligible in all parts except the 0.040 backing plates.

In addition to the models of the repaired panels, a finite element model was also constructed for an unrepaired 1.00 in. hole.

The finite element models indicated that each of the repair concepts should exceed the target strain level of 4,000 μin./in. However, once testing of the specimens began, substantial discrepancies were immediately observed. In particular, the NASTRAN models overestimated the load reacted by the patch while underestimating the strain at the edge of the hole. At the time this was discovered most of the specimens had been fabricated and were scheduled for testing. However, to aid in the design of remaining specimens, a study was undertaken to determine the causes of the discrepancies between predicted and actual behavior.

The study considered the following:

(a) The NASTRAN underestimation of the strain at the edge of the hole

(b) Bolt clearance effects

(c) Bolt stiffness predictions

The underestimation of the strain at the edge of the damage hole was due to two causes. First, the NASTRAN analysis predicts the average strain at the centroid of each element. In the elements around the damage hole no correction was made to determine the difference between the centroidal strain and the maximum strain at the free edge of the element. This difference was later calculated by plotting the centroidal strains of the elements along the centerline of the hole and then extrapolating the curve to the edge of the

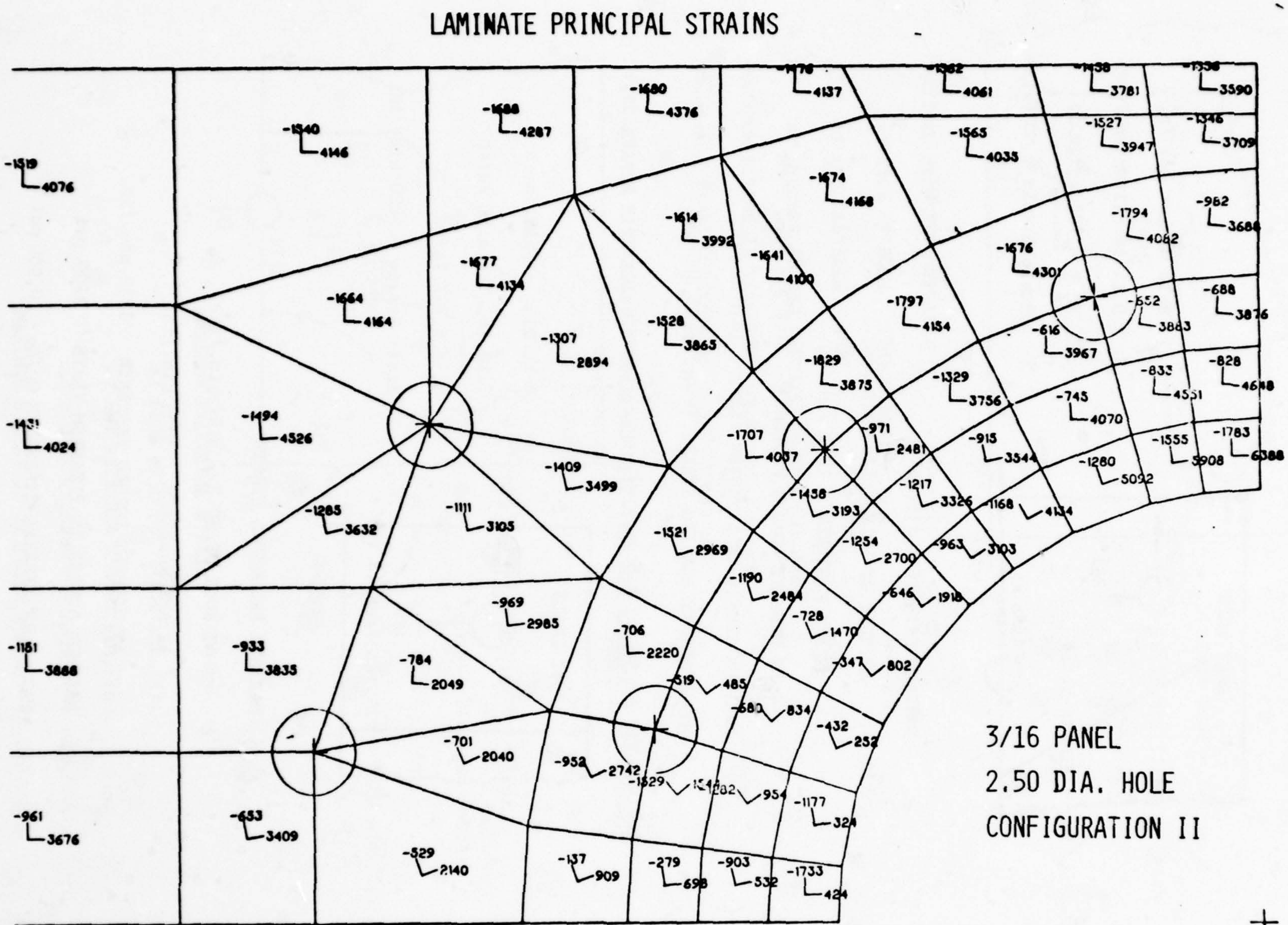
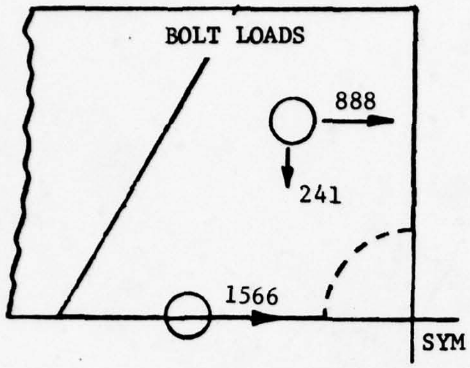


FIGURE 2-2 TYPICAL FINITE ELEMENT REPAIR MODEL



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 6783 μin./in.

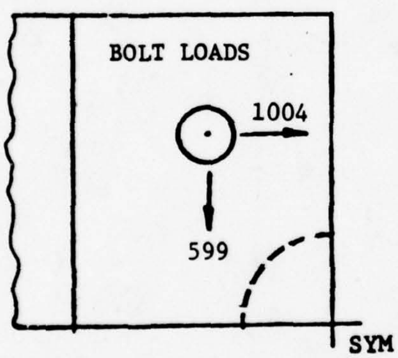
AT ANY FASTENER 4752 μin./in.

AT HIGHEST LOADED FASTENER 3747 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 14,050 psi

MAXIMUM BOLT BEARING STRESS IN Gr/Ep 33,460 psi

FIGURE 2-3 NASTRAN SUMMARY - 3/16 LAMINATE 1.0 DIA HOLE REPAIR #1



Bolt Dia. = 5/16 in.

0.16 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 8147 μin./in.

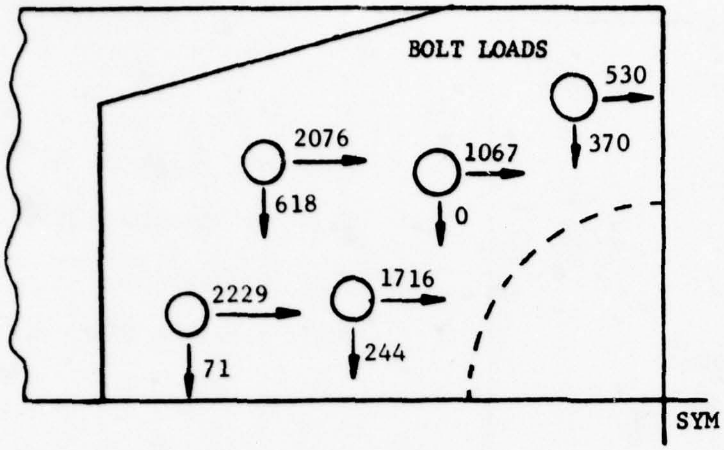
AT ANY FASTENER 5239 μin./in.

AT HIGHEST LOADED FASTENER 5239 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 8500 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 20,000 psi

FIGURE 2-4 NASTRAN SUMMARY-3/16 LAMINATE 1.0 DIA HOLE REPAIR #2



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 6388 μin./in.

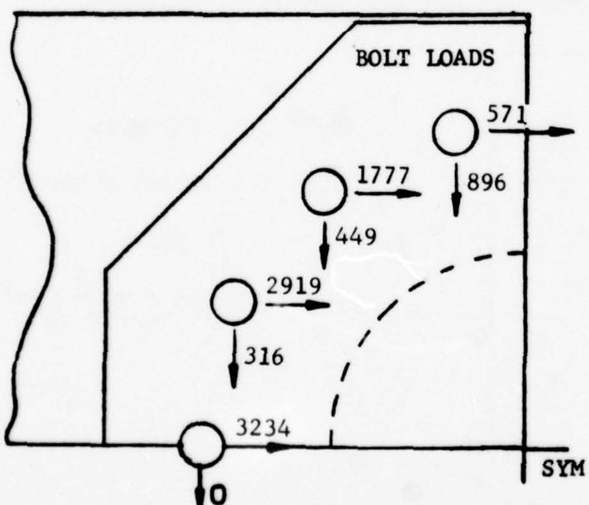
AT ANY FASTENER 4526 μin./in.

AT HIGHEST LOADED FASTENER 3835 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 17,530 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 47,650 psi

FIGURE 2-5 NASTRAN SUMMARY-3/16 LAMINATE 2.5 DIA HOLE REPAIR #1



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 7960 μin./in.

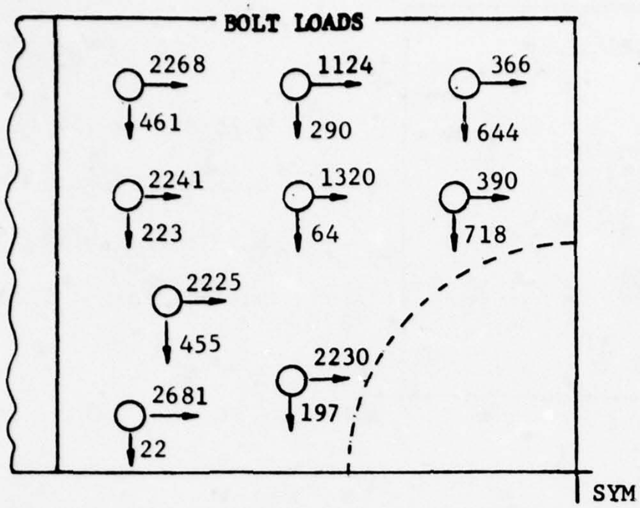
AT ANY FASTENER 5681 μin./in.

AT HIGHEST LOADED FASTENER 3646 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 25,160 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 69,100 psi

FIGURE 2-6 NASTRAN SUMMARY-3/16 LAMINATE 2.5 DIA HOLE REPAIR #2



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 5504 μin./in.

AT ANY FASTENER 4448 μin./in.

AT HIGHEST LOADED FASTENER 3548 μin./in.

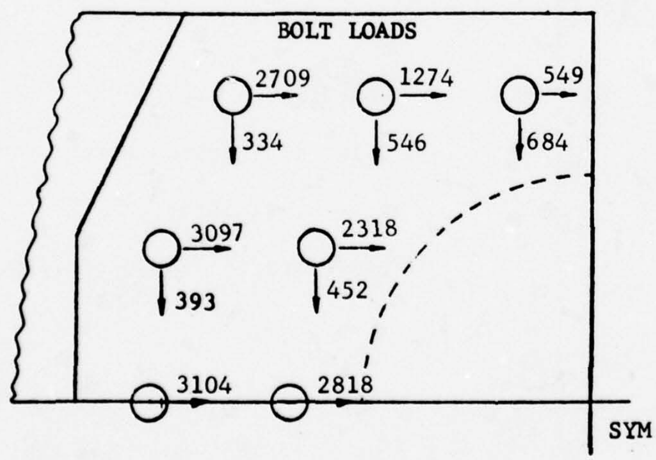
MAXIMUM STRESS IN TITANIUM PLATE 22,906 psi

MAXIMUM BOLT BEARING STRESS IN Gr/Ep 57,286 psi

AVERAGE STRAIN AT HIGHEST LOADED BOLT = 2588 μin./in.

∴ AT 57,286 fbru AV-8B CRITERIA IS MET.

FIGURE 2-7 NASTRAN SUMMARY-3/16 LAMINATE 4.0 DIA HOLE REPAIR #1



Bolt Dia. = 5/16 in.

0.16 in. Titanium Patch

N_X = 6,777 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 5975 μin./in.

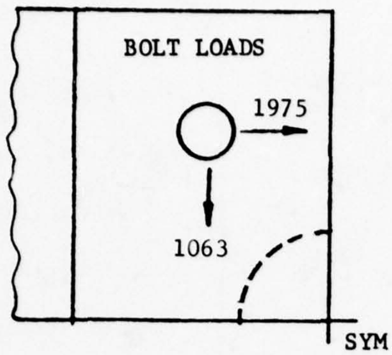
AT ANY FASTENER 5025 μin./in.

AT HIGHEST LOADED FASTENER 3770 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 22,924 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 53,350 psi

FIGURE 2-8 NASTRAN SUMMARY-3/16 LAMINATE 4.0 DIA HOLE REPAIR #2



Bolt Dia. = 5/16 in.

0.16 in. Titanium Patch

0.14 in. Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 8076 μin./in.

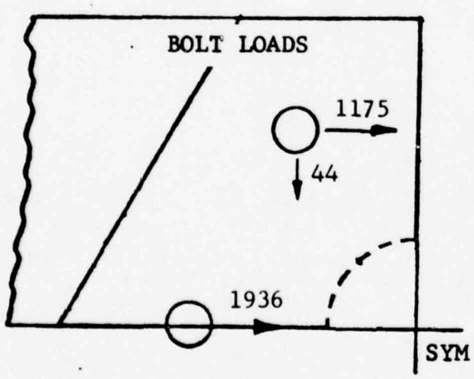
AT ANY FASTENER 5700 μin./in.

AT HIGHEST LOADED FASTENER 5700 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 6888 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 14,350 psi

FIGURE 2-9 NASTRAN SUMMARY-1/2 LAMINATE 1.0 DIA HOLE REPAIR #1



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

0.14 in. Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 7978 μin./in.

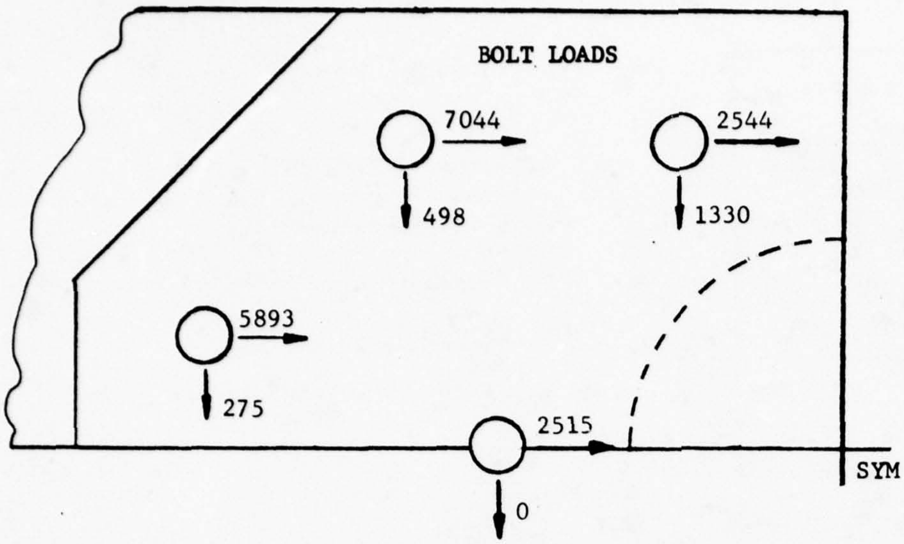
AT ANY FASTENER 4758 μin./in.

AT HIGHEST LOADED FASTENER 3013 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 11,900 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 15,488 psi

FIGURE 2-10 NASTRAN SUMMARY-1/2 LAMINATE 1.0 DIA HOLE REPAIR #2



Bolt Dia. = 5/16 in.

0.16 in. Titanium Patch

0.14 in. Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

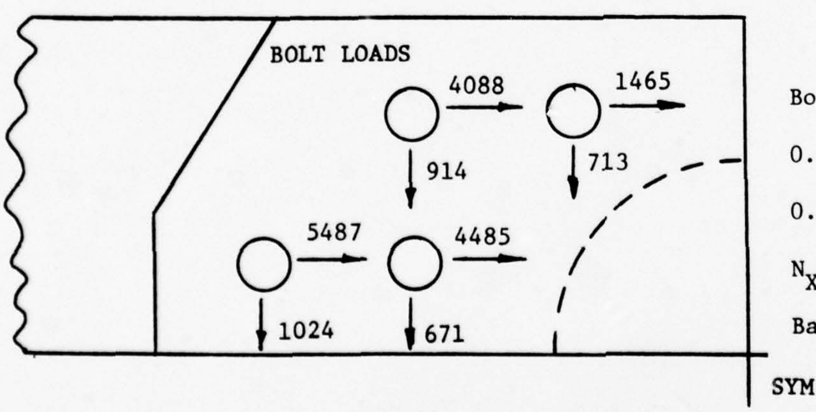
MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 6927 μin./in.
AT ANY FASTENER 5042 μin./in.
AT HIGHEST LOADED FASTENER 4936 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 38,000 PSI

MAXIMUM BOLT BEARING STRESS IN Gr/Ep 45,080 PSI

FIGURE 2-11 NASTRAN SUMMARY-1/2 LAMINATE 2.5 DIA HOLE REPAIR #1



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

0.14 in. Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

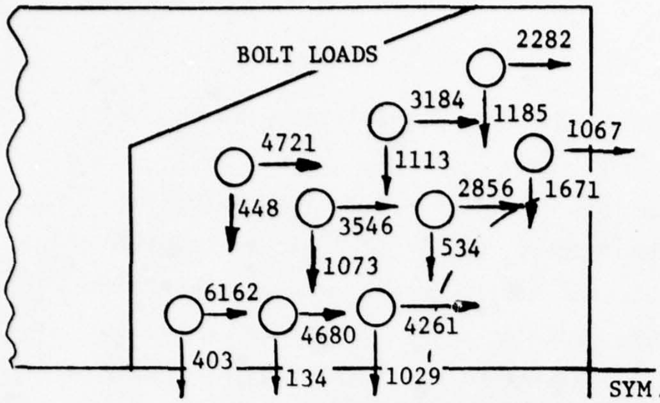
MAXIMUM PRINCIPAL STRAIN -

AT EDGE OF HOLE 9791 μin./in.
AT ANY FASTENER 5334 μin./in.
AT HIGHEST LOADED FASTENER - 5334 μ IN/IN

MAXIMUM STRESS IN TITANIUM PLATE 32,416 PSI

MAXIMUM BOLT BEARING STRESS IN Gr/Ep 44,653 PSI

FIGURE 2-12 NASTRAN SUMMARY-1/2 LAMINATE 2.5 DIA HOLE REPAIR #2



Bolt Dia. = 1/4 in.

0.14 in. Titanium Patch

0.14 in Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN

AT EDGE OF HOLE 7639 μin./in.

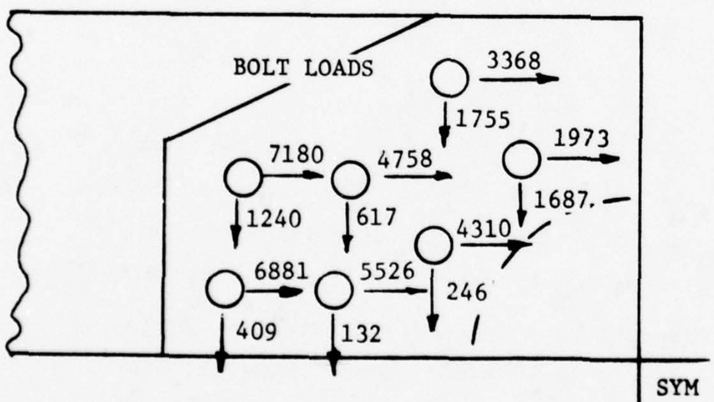
AT ANY FASTENER 5190 μin./in.

AT HIGHEST LOADED FASTENER 3737 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 54,304 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 51,565 psi

FIGURE 2-13 NASTRAN SUMMARY-1/2 LAMINATE 4.0 DIA HOLE REPAIR #1



Bolt Dia. = 5/16 in.

0.16 in. Titanium Patch

0.14 in. Titanium Backing Plate

N_X = 19,150 lb/in.

Basic Strain = 4000 μin./in.

MAXIMUM PRINCIPAL STRAIN -

AT EDGE OF HOLE 7527 μin./in.

AT ANY FASTENER 5658 μin./in.

AT HIGHEST LOADED FASTENER 4302 μin./in.

MAXIMUM STRESS IN TITANIUM PLATE 41,632 psi

MAX. BOLT BEARING STRESS IN Gr/Ep 49,172 psi

FIGURE 2-14 NASTRAN SUMMARY-1/2 LAMINATE 4.0 DIA HOLE REPAIR #2

hole. These corrections typically resulted in about a 25% increase in the maximum strains shown in Figure 2-3 thru 2-14.

The second reason for underestimation resulted from the coarseness of the finite element mesh surrounding the cutout. This could not be resolved without major changes to the finite element model. As an example, the finite element model for an unrepaired 1.00 in. hole in a 3/16 in. laminate predicts a strain concentration factor at the edge of the hole of 3.04. However, the exact theoretical strain concentration for the same hole in an infinitely-wide laminate is 3.50, using the solution of Reference (3). Multiplying this times a finite width correction factor of 1.20, Reference (4), the theoretical strain concentration factor is 4.20, or 38% greater than initially predicted by the NASTRAN finite element model.

Fastener clearance effects can also lead to differences between the predicted and actual performance. This is because the laminate must be preloaded to a certain strain level before the fasteners become effective in transferring load from the laminate to the repair patch. NASTRAN cannot handle the nonlinearity created by gapping between the hole boundary and the fastener. Consequently, NASTRAN tended to overestimate the effectiveness of the repair patch.

Bolt stiffness is important in determining the effectiveness of the repair concept. Therefore, it is vital for an exacting analysis of the specimens, to have a method for predicting the bolt stiffness for the different combinations of laminate and patch thickness, fastener material and diameter, fastener clamp-up, etc., for the single shear or double shear loadings that mav occur.

In previous NASTRAN finite element computations, the empirical equation of Voight, Ref. (2), was used for computing bolt stiffness. However, the Voight equations did not distinguish between different bolt materials, predicting identical results for both steel or aluminum fasteners. It also did not account for different head clamping conditions.

To determine the role of these parameters, additional bolt stiffness computations were made, using a more complete method Ref. (5). Table 2-1 summarizes these predictions, along with those obtained using Voight's method, for the following cases:

	Fastener Dia. Inches	Laminate Thickness Inches	BOLT STIFFNESS - K_{bolt} (10^{-6} lb/in.)					
			Titanium			Steel		
			Clamped Head △1	Free Head △1	△2	Clamped Head △1	Free Head △1	△2
Single	.250	.189	.538	.226	.516	.688	.241	.516
		.50	.519	.405	.731	.664	.400	.731
Shear	.3125	.189	.666	.239	.600	.813	.242	.600
		.50	.673	.463	.862	.862	.491	.862
Double	.250	.189	1.008	.927	.966	1.111	1.050	.966
		.50	1.153	.842	1.222	1.427	1.069	1.222
Shear		.189	1.135	1.078	1.214	1.219	1.179	1.214
	.3125	.50	1.479	1.137	1.419	1.762	1.431	1.419

△1 **Computed using method of Ref. (5)**

△2 **Computed using method of Ref. (2)**

TABLE 2.1 **COMPARISON OF BOLT STIFFNESSES**

- Single and double shear
- 3/16 in. and 1/2 in. laminates
- Steel and titanium fasteners
- 1/4 in. and 5/16 in. dia fasteners
- Clamped and unclamped heads

The data shows the following trends

- For single shear fasteners, with unclamped heads, steel and titanium fasteners have the same stiffness
- Fasteners in double shear are considerably stiffer than those in single shear.
- A modest increase in stiffness is possible by using steel fasteners for double shear applications. The same increase can be achieved by use of 5/16 in. fasteners in lieu of 1/4 in.

Table 2.1 shows, for the most part, no dramatic differences in the predictions obtained using the more complete methods. However, the stiffness values in Table 2.1 for the case of unclamped heads were used for all subsequent analyses to keep predictions conservative.

The initial strain criterion of 10,000 μin./in. at the hole edge was also reevaluated. This was because such a criterion did not consider the role of hole size in degrading strength. Tensile failure occurs when the strain at the edge of the hole exceeds a critical level. The level of this hole strain depends upon diameter, because of the nonlinear material behavior in a "damage" zone at the hole edge.

The characteristic dimension hypothesis, presented in Ref. (6), assumes that a zone of damaged material forms at the edge of the hole prior to failure. The size of the damage zone is assumed to be a material property and is constant for all diameter holes. Small damage holes therefore have proportionately larger damage zones than large holes. The damage consists of local micro delaminations, fiber failure and matrix micro cracking. This has the effect of yielding the material and therefore eliminates peaking of the hoop tensile stress at the edge of the hole. The stress level remains essentially constant from the edge of the hole to the interface between damaged and undamaged laminate. Failure occurs when the stress state in this area exceeds a critical value.

The tangential stress at a distance (x) from the edge of the hole is approximated using the solution for an open hole in an isotropic plate, Ref. (9):

$$\sigma_x = \sigma_o \quad 1 + \frac{1}{2}\left(\frac{R}{R+a}\right)^2 + \frac{3}{2}\left(\frac{R}{R+a}\right)^4$$

where: R = hole radius

a = characteristic dimension

σ_o = far-field stress

The critical stress level, at which failure occurs, is invariant for a given material system and ply stacking sequence. Therefore, the gross failure strain of a laminate containing a hole of radius R can be estimated using the results of a baseline damaged laminate:

$$\varepsilon = \varepsilon^* \left[\frac{(1 + \frac{1}{2}\left(\frac{R^*}{R^*+a}\right)^2 + \frac{3}{2}\left(\frac{R^*}{R^*+a}\right)^4}{(1 + \frac{1}{2}\left(\frac{R}{R+a}\right)^2 + \frac{3}{2}\left(\frac{R}{R+a}\right)^4} \right]$$

where: R* = hole radius in baseline laminate

ε^* = gross failure strain of baseline laminate

a = characteristic dimension

Figure 2-15 shows the expected strength reduction for AS/3501-6 graphite/epoxy. This curve shows good agreement with available test results from Ref. (1), and shows that for the hole sizes considered, strength reduction between 25% and 50% occur. The average unloaded hole tension failing strain for laminates containing 1/4" diameter holes in AS/3501-6 layups, typical of the AV-8B torque box skins, has been found to be 5200 μin./in. This value can be used to determine the permissible strain at the hole edge for various diameters.

The corresponding strain at the hole edge can be estimated by multiplying 5200 μin./in. by the stress concentration factor for the tensile coupon specimen. Using the equation of Ref. (3) for an open hole in an infinite orthrotropic plate, along with the finite width correction factor of Ref. (4), the failure strain at the hole edge was found to be 21,840 μin./in.

Using this baseline strength in conjunction with the reduction factors shown in Figure 2-15, the maximum strain levels at the edge of the hole were computed and are shown in Table 2.2.

A simple analytical method was developed for redesign of the remaining specimens. The method was based on the "compliance" concept, Ref. (7), and accounts for the nonlinearity created by fastener clearance in addition to providing accurate estimates of the strain at the hole edge. Failure was predicted when the hole strain exceeded the maximum value shown in Table 2.2.

In the "compliance" approach the in-plane shear lag in both the skin and any reinforcing members was neglected. Also, it was assumed that bolt loads

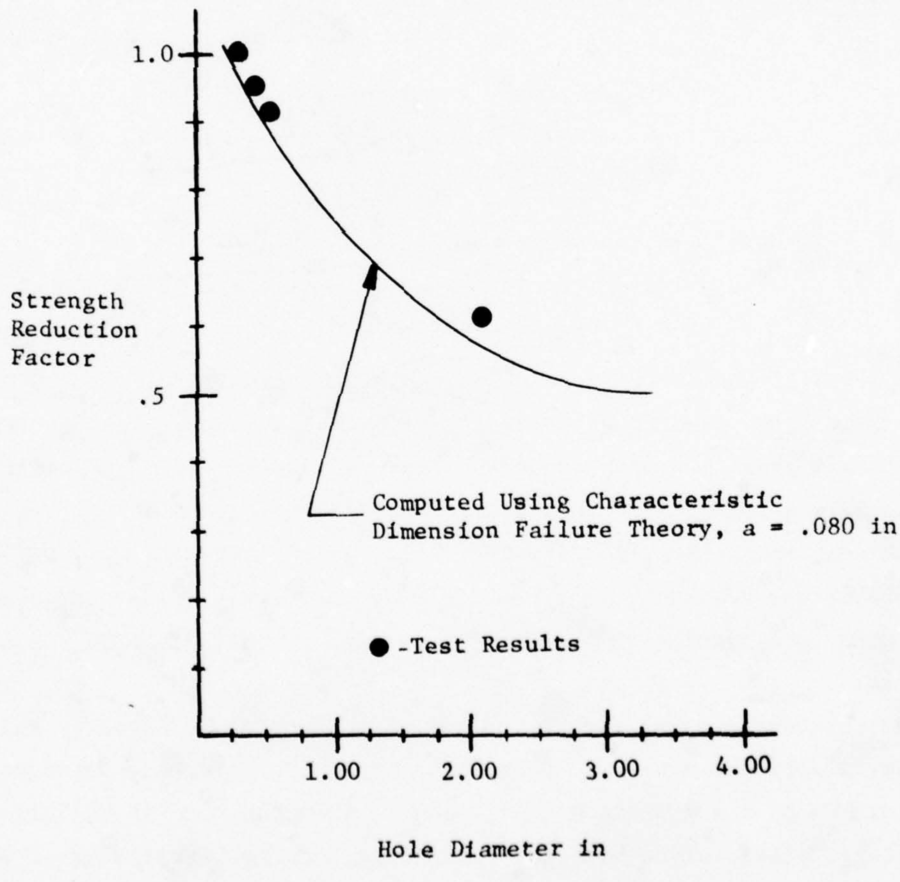


FIGURE 2-15 HOLE SIZE VS STRENGTH REDUCTION FACTOR FOR GR/EP LAMINATES

at a point produce a state of uniform axial stress and deformation across the section. In this manner, it was possible to idealize the-laminate and repair patch as one-dimensional bar members interconnected by flexible fasteners.

Complete details of the approach are presented in Appendix C. Section 4 shows that this method agrees well with test results.

Table 2.3 summarizes the predicted strain at the edge of the hole, the maximum bearing stress in the laminate at a gross laminate strain at failure for all repairs tested, including the follow-on test configurations. In all computations, fastener clearance of 0.002 in. was assumed. Frictional clamp-up between the panel and patch was ignored. In addition, the earlier ground-rule of keeping the maximum bearing stress in the panel below 50,000 psi was relaxed in an effort to increase the load transfer into the repair patches.

No analysis was attempted on the off-axis specimens, because the compliance method is not suitable when coupling is exhibited between inplane shear and extensional deformations.

Hole Diameter (in.)	Predicted Hole Edge Failure Strain μin./in.
.25	21,840
1.00	16,380
2.50	11,575
4.00	11,138

TABLE 2.2 PREDICTED MAXIMUM HOLE STRAIN VS. HOLE DIAMETER

SPECIMEN	HOLE DIA INCH	PANEL THICKNESS INCH	PREDICTED HOLE STRAIN @ ε_o=4,000 μin./in. μin./in.	PREDICTED MAX. BEARING STRESS @ ε_o=4,000 μin./in. psi	PREDICTED GROSS STRAIN AT FAILURE (1) μin./in.
75T060105-1001	1.00	3/16	16,632	1,358	3,900
-1003	1.00	↑	15,237	23,132	4,299
-1005	2.50		11,430	39,894	4,051
-1007	2.50		13,320	38,540	3,476
-1009	2.50		(2)	(2)	(2)
-1011	4.00		11,160	50,540	3,992
-1013	4.00		9,540	45,439	4,670
-1015	4.00		8,640	61,715	5,156
-1017	2.50	↓	(2)	(2)	(2)
-1019	1.00	3/16	10,440	85,630	6,724
75T060106-1001	1.00	1/2	16,750	598	3,910
-1003	1.00	↑	16,212	9,552	4,041
-1005	2.50		11,593	61,114	4,019
-1007	2.50		13,320	50,288	3,476
-1009	4.00		10,265	55,718	4,670
-1011	4.00		10,374	59,200	4,295
-1013	2.50		(2)	(2)	(2)
-1015	2.50		(2)	(2)	(2)
-1017	2.50		11,340	62,440	4,083
-1019	1.00	↓	13,608	34,721	4,815
-1021	1.00	1/2	15,372	23,500	4,285

(1) BASED ON HOLE FAILURE
(2) NOT ANALYZED BY COMPLIANCE MODEL APPROACH

TABLE 2.3 COMPLIANCE MODEL ANALYSIS SUMMARY

3. TESTING

Initial tests were conducted to establish which of the two repair concepts performed best in each damage configuration. Each concept was tested in static tension. Subsequent tests were conducted to substantiate repeatability of the better repairs, to evaluate off-axis sensitivity and to determine compression properties.

3.1 TEST SETUP AND PROCEDURES

3.1.1 TENSION TESTS - All tension specimens for this program were tested on a 400,000 lb Baldwin-Lima Universal Test Machine. Load plates were bolted to each end of the repair specimens and attached to the test machine with a single-pin joint at each end. Figure 3-1 shows a typical tension test setup.

Test specimens were strain gaged in accordance with the 75T060105 and 75T060106 assembly drawings, Figures A-5 and A-6 in Appendix A. Figure 3-2 shows the general arrangement and numbering of the strain gages on the initial specimens.

After initial tests, some changes were made on the instrumentation. The rosette gage was replaced on the center of the patch with an axial gage, and the other two strain gages were eliminated. See Figure 3-3. These extra gages had been intended to record any uneven load distribution. Initial tests however, showed that no significant uneven loading occurs. Some of the re-designed patches used in the follow-on tests did not allow room for the six axial gages used to measure gross laminate strain. On these specimens, two gages were mounted on the edges of the laminate as shown in Figure 3-4. All strain gage data presented later in this report will be identified as shown in Figures 3-2, 3-3, and 3-4.

The specimens were loaded in 10% Test Limit Load (TLL) increments until failure. The load rate was .05 in./min. Strain data was recorded at each increment until the gross laminate strain reached 3000 μin./in., at which time data was recorded continuously to failure.

During selected tests in the follow-on test phase, data was also recorded continuously from 0 to the first load increment. Table 3.1 lists the TLL and test load increments for each type of specimen.

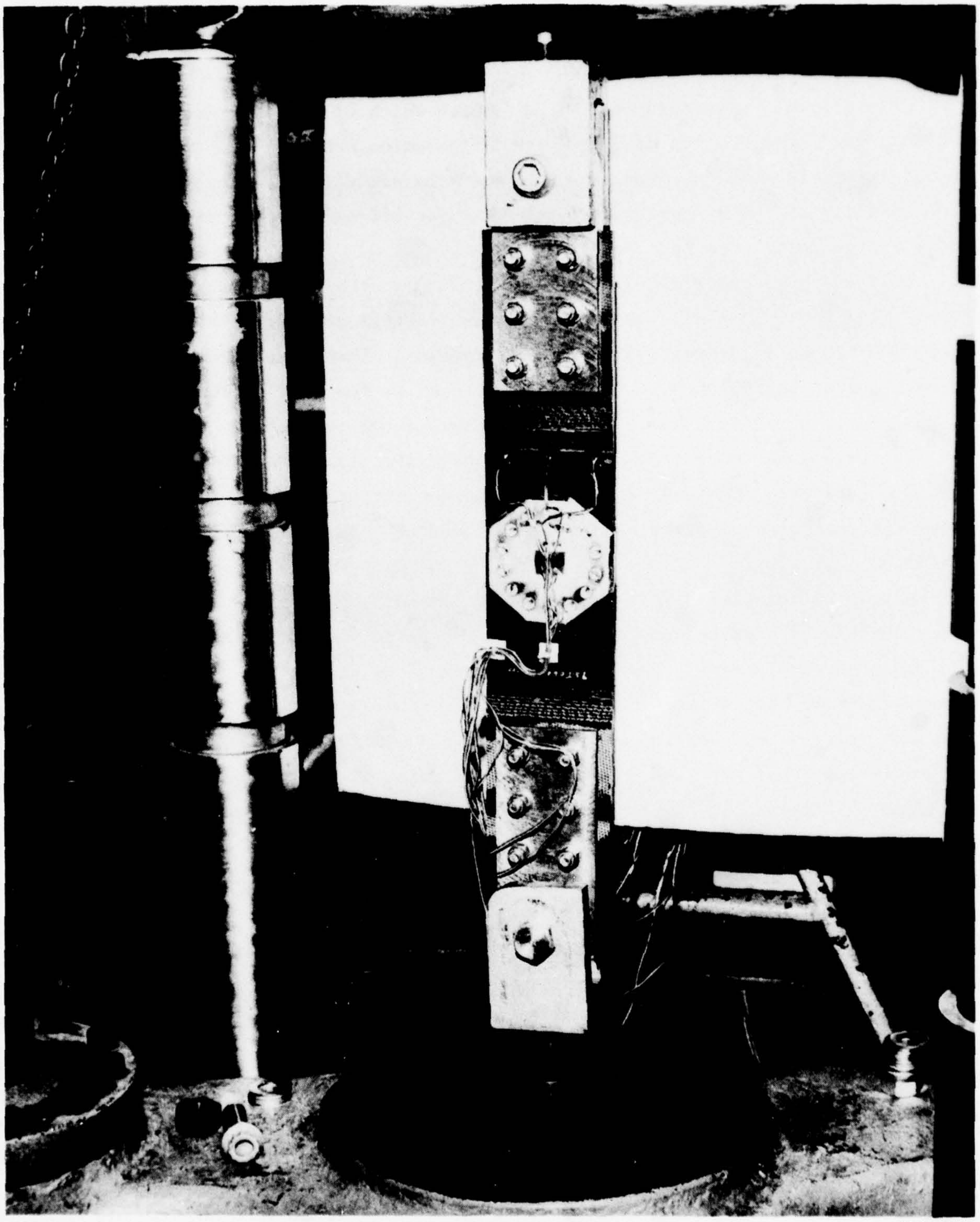


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FIGURE 3-1 TYPICAL TENSION TEST SETUP

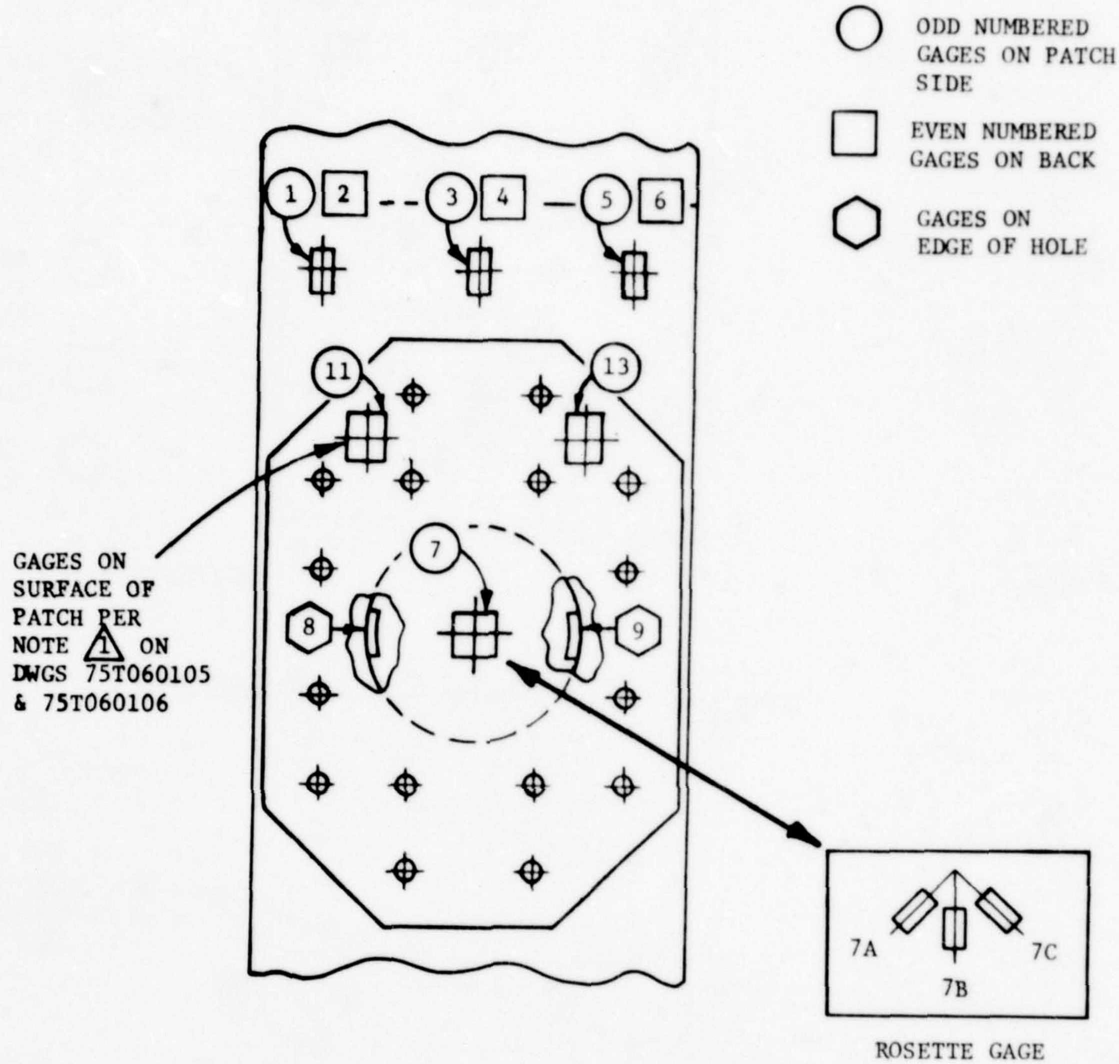

FIGURE 3-2 STRAIN GAGE INSTRUMENTATION INITIAL TESTS

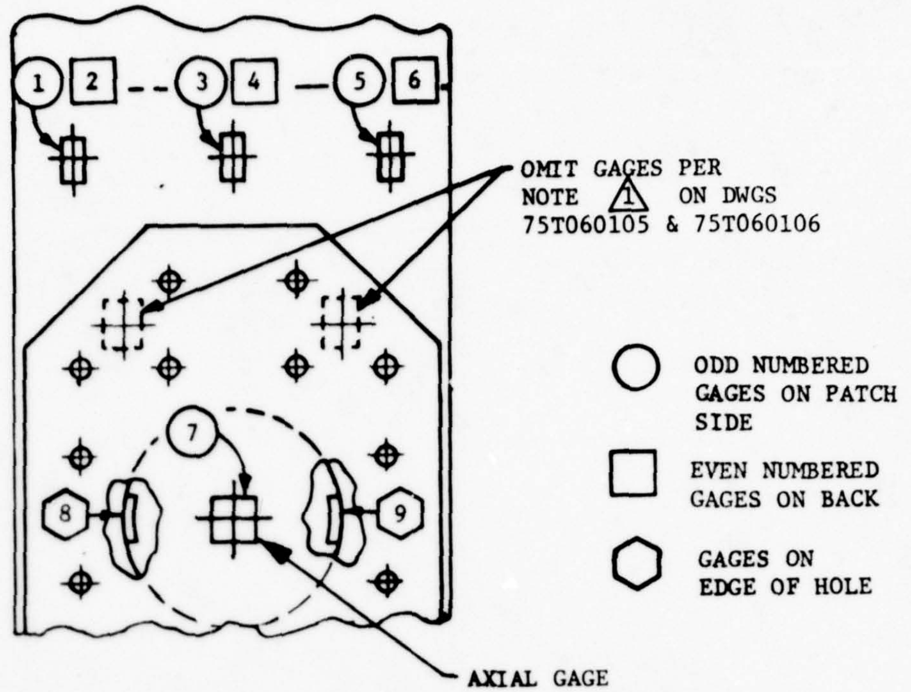

FIGURE 3-3 SIMPLIFIED STRAIN GAGE INSTRUMENTATION

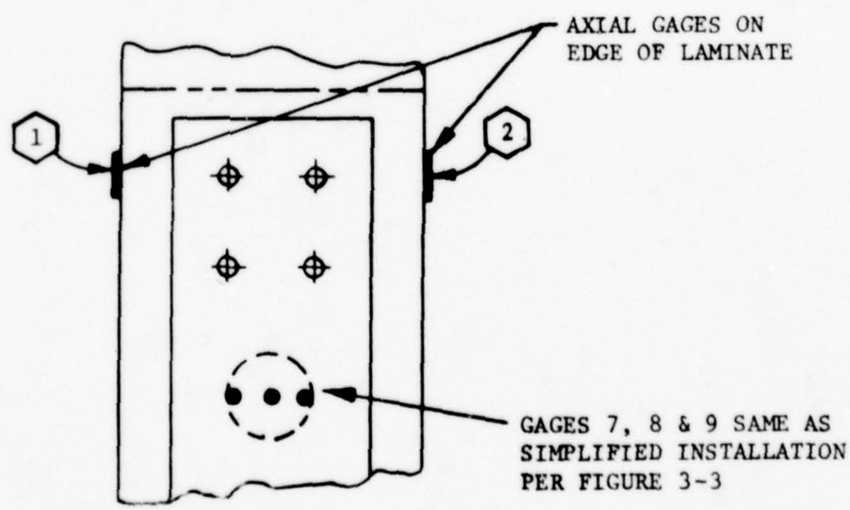

FIGURE 3-4 ALTERNATE SIMPLIFIED STRAIN GAGE INSTRUMENTATION

Specimen		Load Increment (lb)	100% TLL (lb)
3/16 Laminates	1.0 Dia Hole	3,000	29,645
	2.5 Dia Hole	4,000	42,350
	4.0 Dia Hole	7,000	67,760
1/2 Laminates	1.0 Dia Hole	8,000	83,720
	2.5 Dia Hole	12,000	119,600
	4.0 Dia Hole	20,000	191,360

TABLE 3.1 TEST LOAD INCREMENTS FOR TENSION TESTS

3.1.2 COMPRESSION TESTS - The compression tests were performed in the same Baldwin-Lima test machine used for the tension tests.

The compression panels were loaded through blocks bonded to the ends of the specimens (see Figures A-5 and A-6 in Appendix A). The edges of the specimens were restrained by supports fabricated from steel pipe. Each pipe had a slot cut along its length in which the edge of the specimen edge was inserted. Bolts through the pipe provided clamping force to support the specimen edge securely. Drawing 75T060107 (Figure A-7, Appendix A) defines the pipe edge supports. Figure 3-5 is a photograph of a typical compression test setup.

Figure 3-3 shows the configuration of the strain gage instrumentation used on the compression specimens. The test procedure called for loading the 3/16 in. laminate in 7,000 lb increments, and the 1/2 in. laminate in 20,000 lb increments until an average laminate strain of 4,000 μin./in. was reached or until the onset of panel buckling. Data was recorded at each increment.

The objective of these tests was to demonstrate the compression characteristics of the repair patches without damaging the specimens. The specimens are scheduled for further testing by NADC.

3.2 TESTING OF 3/16 INCH LAMINATES WITH 1.0 DIA. DAMAGE HOLE

The two initial repair designs tested for 3/16 in. laminates with 1.0 in. holes are shown in Figure 3-6. These specimens are fully defined in the

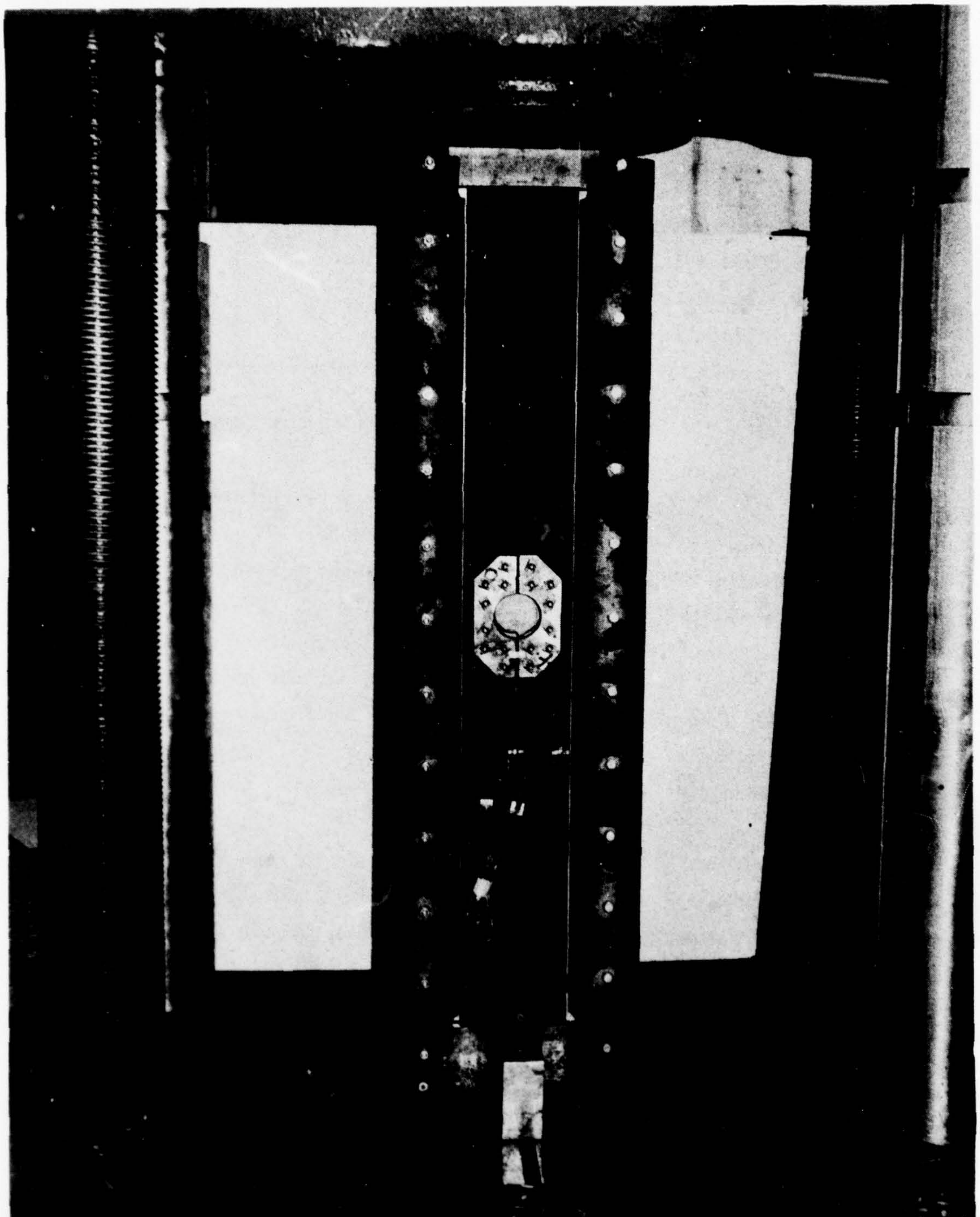

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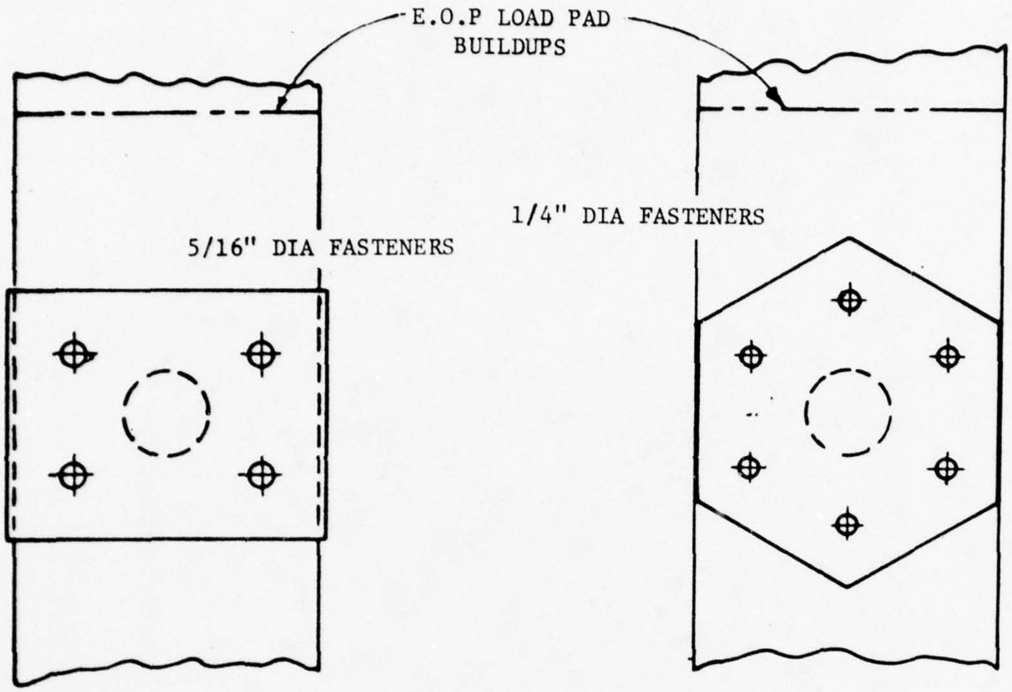
FIGURE 3-5 TYPICAL COMPESSION TEST SETUP



75T060105-1001 ASSEMBLY

75T060105-1003 ASSEMBLY

Part Name	Drawing Part Number 4-Bolt Configuration	 6-Bolt Configuration
Repair Assy	75T060105-1001	75T060105-1003
Patch Details	75T060103-2011/-1013	75T060103-2009/-1011
Gr/Ep Specimen	75T060101-1001	75T060101-1001

FIGURE 3-6 3/16 LAMINATE 1.0 DIA DAMAGE HOLE INITIAL DESIGNS

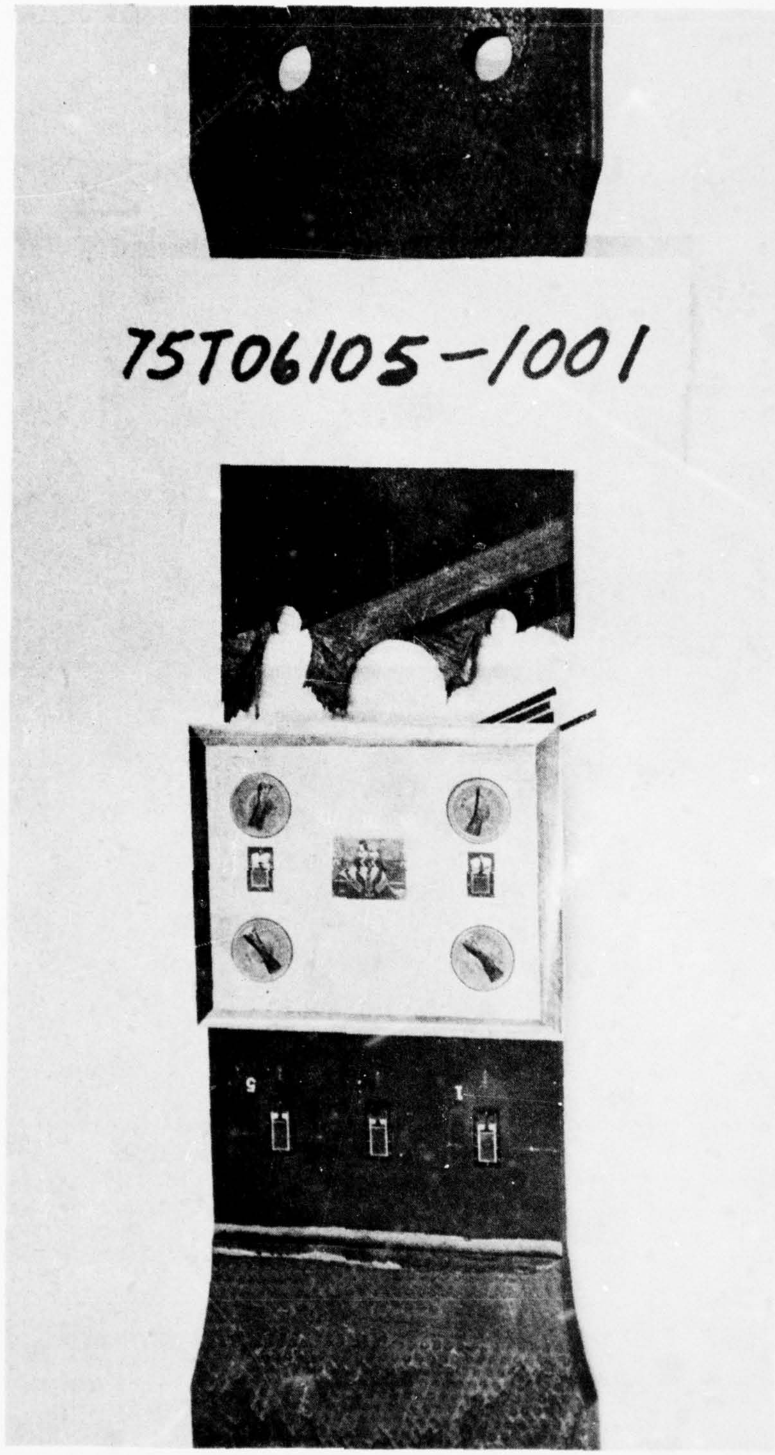


FIGURE 3-7
FAILED 75T060105-1001 SPECIMEN
3/16 Laminate 1.0 Diameter Damage Hole

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Engineering drawings Figures A-1, A-3, and A-5 in Appendix A.

The 75T060105-1001 (4 bolt) repair failed at 23,300 lb producing 3,846 μin./in. average laminate strain. The specimen failed in the net section across the damage hole and the laminate failed around the patch fasteners as shown in Figure 3-7.

Gages on the surface of the patch recorded 30 μin./in. of strain at failure while strain at the edge of the hole was measured at 19,914 μin./in. A complete set of strain gage data is presented in Appendix B, Figure B-1.

The 75T060105 (6 bolt) repair failed at a total load of 25,000 lb. The average strain in the laminate was 4,039 μin./in. The strain at the edge of the hole was only recorded to 10,000 μin./in. due to limits set in the data recording system. (This condition was corrected on later tests). The maximum strain recorded on the patch was 200 μin./in.

The specimen failed in the net section across the damage hole. The Gr/Ep laminate did not fail around the fasteners after the net section failed. The specimen remained in one piece with the patch holding it together, as shown in Figure 3-8. See Appendix B, Figure B-2 for strain gage data.

These two repairs did not perform as predicted by the NASTRAN analysis. Consequently, following these tests the analytical methods were reviewed. This review and resulting analytical revisions are discussed in Section 2.

Following the review, two new repairs were designed for the 1.0 dia. damage hole in both the 3/16 and 1/2 in. laminates. The initial intention was to test both configurations in the 1/2 in. laminate and choose the best repair for follow-on testing for both laminates. Unfortunately, the schedule did not permit this, and analysis was used to select the least-risk path on the 3/16 laminates.

Figure 3-9 shows the revised configuration tested on the final two 3/16 laminates. The table included in that figure lists the parts which comprise the specimen assembly (Ref Appendix A, Figures A-1, A-4, and A-5).

The new 8-bolt configuration was a significant improvement. Failures at 37,000 and 35,600 lb produced average laminate strains of 5,712 and 5,604 μin./in., respectively. Strains at the edge of the damage hole were 15,175 and 13,083 μin./in. Patch strains were 875 and 980 μin./in. Failure came along the outer bolt line at one end of each specimen. See Figure 3-10.

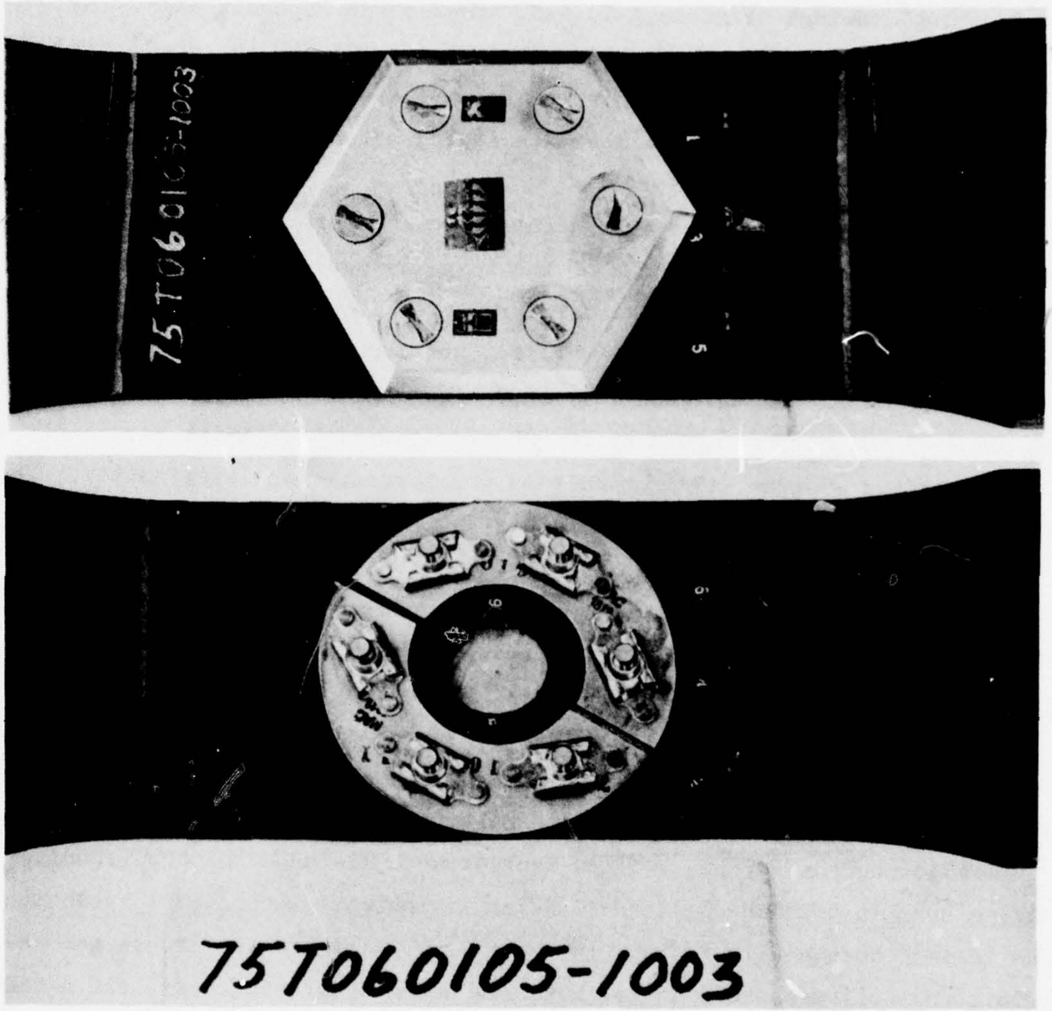

FIGURE 3-8
FAILED 75T060105-1003 SPECIMEN
3/16 Laminate 1.0 Diameter Damage Hole

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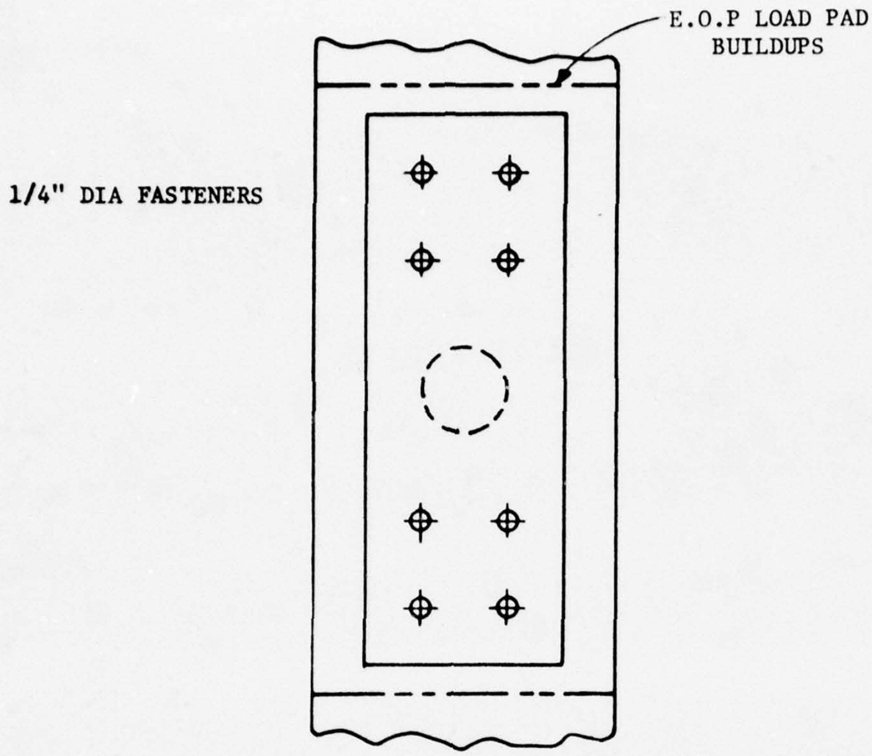
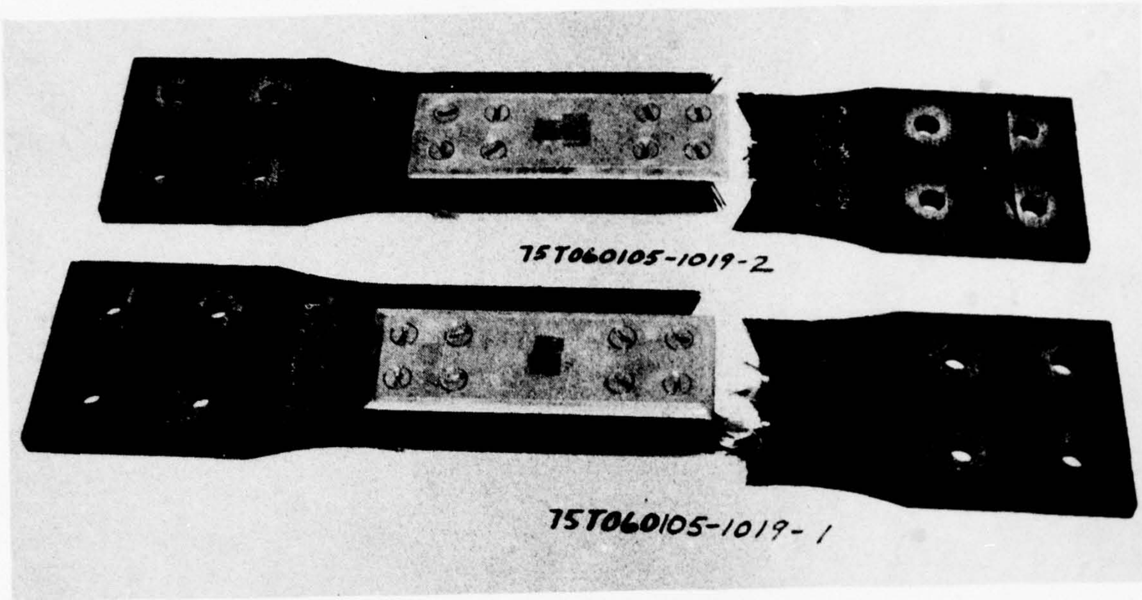


75T060105-1019 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060105-1019
Patch Details	75T060104-2011/-1019
Gr/Ep Specimen	75T060101-1001

FIGURE 3-9 3/16 LAMINATE 1.0 DIA DAMAGE HOLE
FOLLOW-ON TEST DESIGN



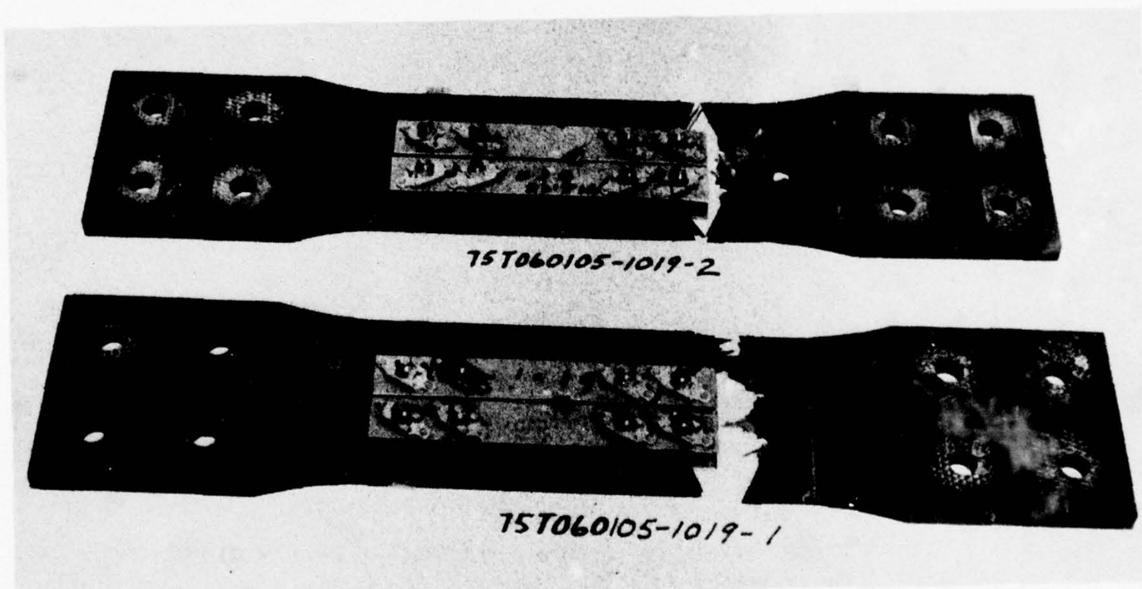


FIGURE 3-10
FAILED 75T060105-1019 SPECIMENS
3/16 Laminate 1.0 Diameter Damage Hole

GP78-8701-16

See Figures B-3 and B-4, Appendix B, for complete gage data. Table 3.2 summarizes the test data.

SPECIMEN 75T060105	FAILURE LOAD		STRAIN AT FAILURE μin./in.			STRAIN RATIO
	lb	lb/in.	LAMINATE	HOLE	PATCH	PATCH/LAMINATE
-1001	23,300	6,714	3846	19,914	30	.01
-1003	25,500	7,142	4039	10,000+	200	.05
-1019 #1	37,000	10,571	5712	15,175	875	.15
-1019 #2	35,600	10,171	5604	13,083	980	.17

TABLE 3.2 SUMMARY OF TEST DATA - 3/16 LAMINATE - 1.0 DIA. DAMAGE HOLE

3.3 TESTING OF 3/16 INCH LAMINATES WITH 2.50 DIA. DAMAGE HOLE

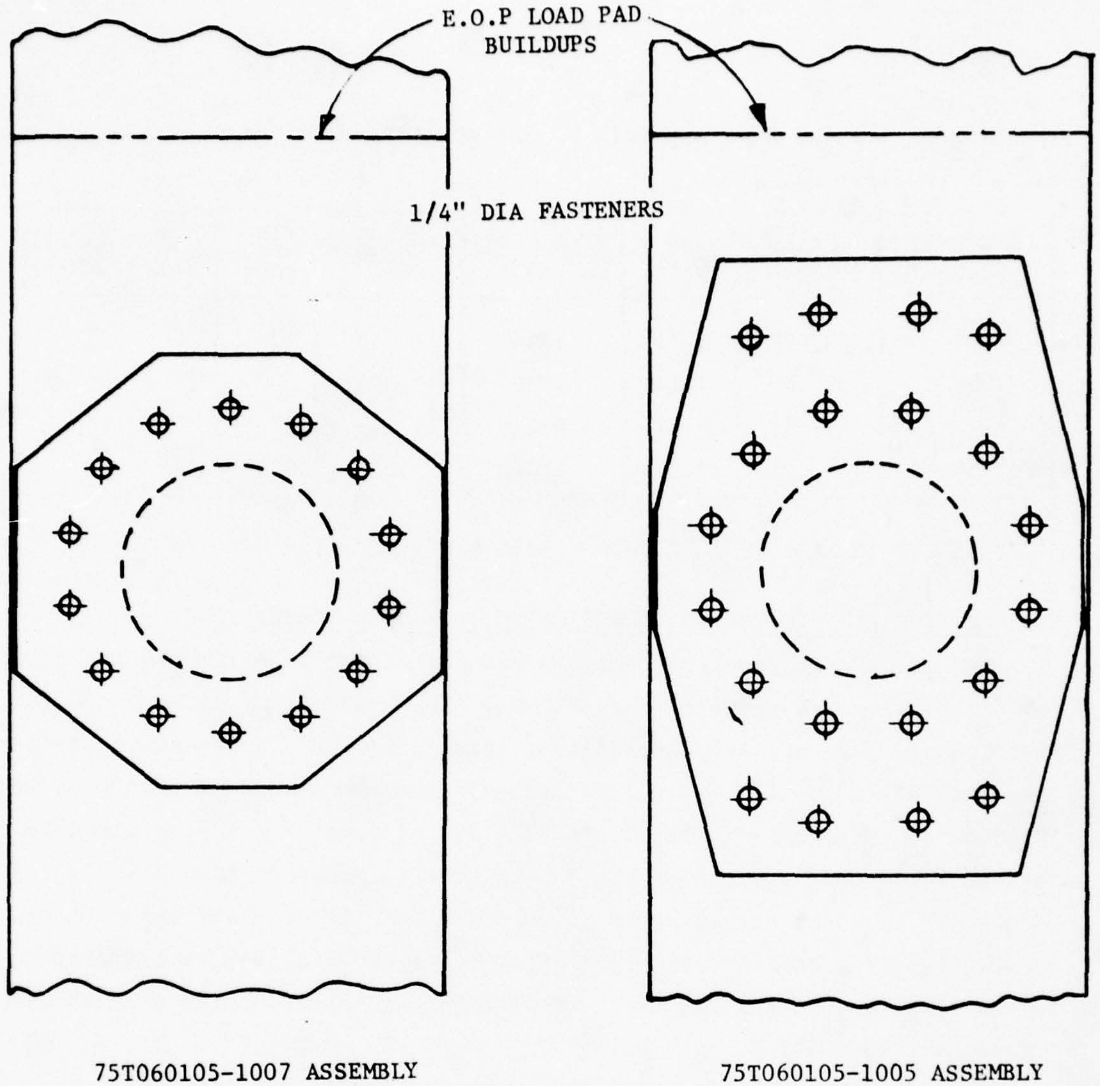
The two repair concepts designed and tested for 3/16 inch laminates with 2.50 inch holes are shown in Figure 3-11. These specimens are defined in Figures A-1, A-3, and A-5, Appendix A. Both concepts use 1/4 in. fasteners.

The 75T060105-1007 (circular pattern) specimen failed at 36,460 lb which produced an average laminate strain of 4,291 μin./in. It failed along the fastener line near one end of the damage hole as shown in Figure 3-12.

At failure the strain at the edge of the damage hole was measured at 13,605 μin./in., and the patch was strained to 873 μin./in. at the center in the direction of the loading. Complete strain gage data is presented in Figure B-6, Appendix B.

The 75T060105-1005 (rectangular pattern) repair specimen failed at 46,000 lb, with 5,316 μin./in. average laminate strain. Failure occured along the outermost row of fasteners at one end of the specimen (see Figure 3-13). At failure, the patch strain was measured at 1,569 μin./in. at the center. The strain at the edge of the laminate hole was in excess of 10,000 μin./in. (Exact strain value was undetermined because the strain range was set at 10,000 μin./in. in the data recording equipment). Complete strain gage data is presented in Appendix B, Figure B-5.

Both patch designs produced acceptable test results. The 75T060105-1007 has merit due to the fewer number of fasteners and simple fastener pattern. However the 75T060105-1005 was chosen for follow-on testing because it carried a higher load at failure. The two additional specimens failed at 42,100 and



Part Name	Drawing Part Number 14 Bolt Circular Pattern	20 Bolt Rectangular Pattern
Repair Assy	75T060105-1007	75T060105-1005
Patch Details	75T060103-2003/-1003	75T060103-2001/-1001
Gr/Ep Specimen	75T060101-1003	75T060101-1003

FIGURE 3-11 3/16 LAMINATE 2.5 DIA DAMAGE HOLE REPAIR DESIGNS

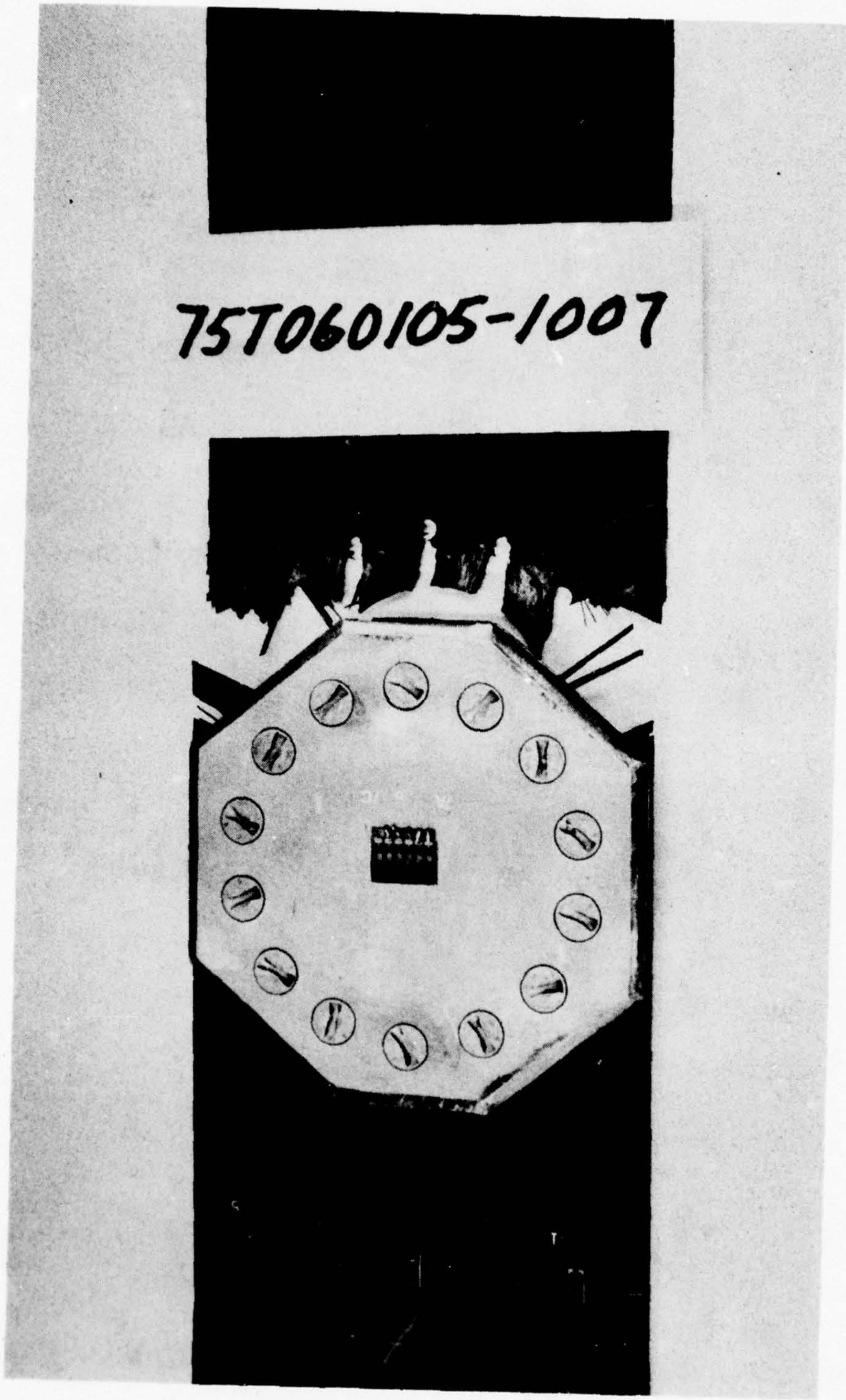


FIGURE 3-12
FAILED 75T060105-1007 SPECIMEN
3/16 Laminate 2.5 Diameter Damage Hole

GP78-8701-8

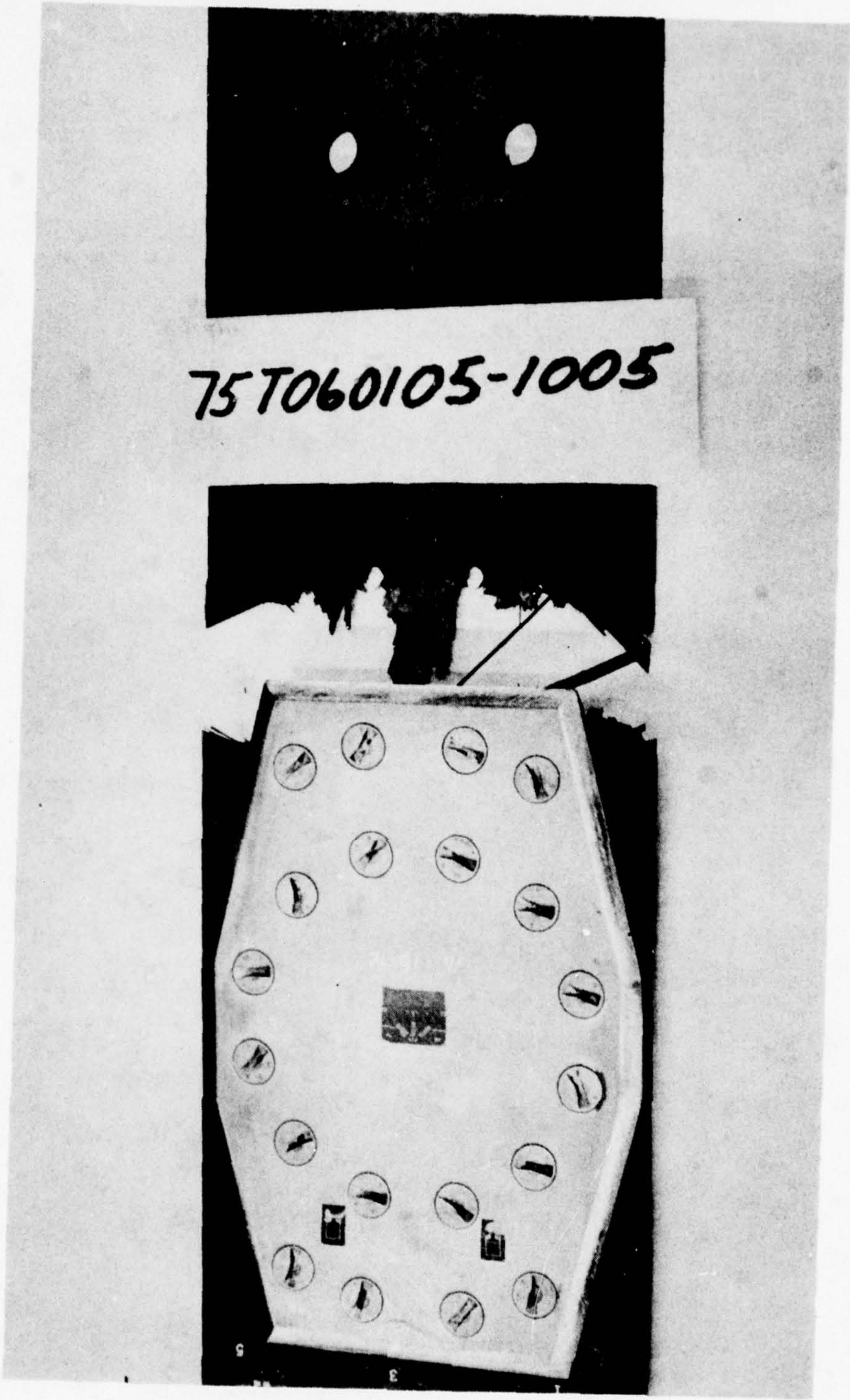


FIGURE 3-13
FAILED 75T060105-1005 SPECIMEN
3/16 Laminate 2.5 Diameter Damage Hole

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46,600 lb, which yield strains of 4,893 μin./in. and 5,358 μin./in., respectively (See Appendix B, Figure B-7 and B-8). The failure modes were identical to the first specimen. These tests substantiate the structural integrity of the repair configuration. Table 3.3 summarizes the test information.

Specimen 75T060105	Failure Load		Strain at Failure			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1007	36,460	7250	4291	13,605	873	.20
-1005 #1	46,000	9220	5310	10,000+	1569	.30
-1005 #2	42,100	8420	4893	-	1378	.28
-1005 #3	46,600	9320	5358	15,441	1485	.28

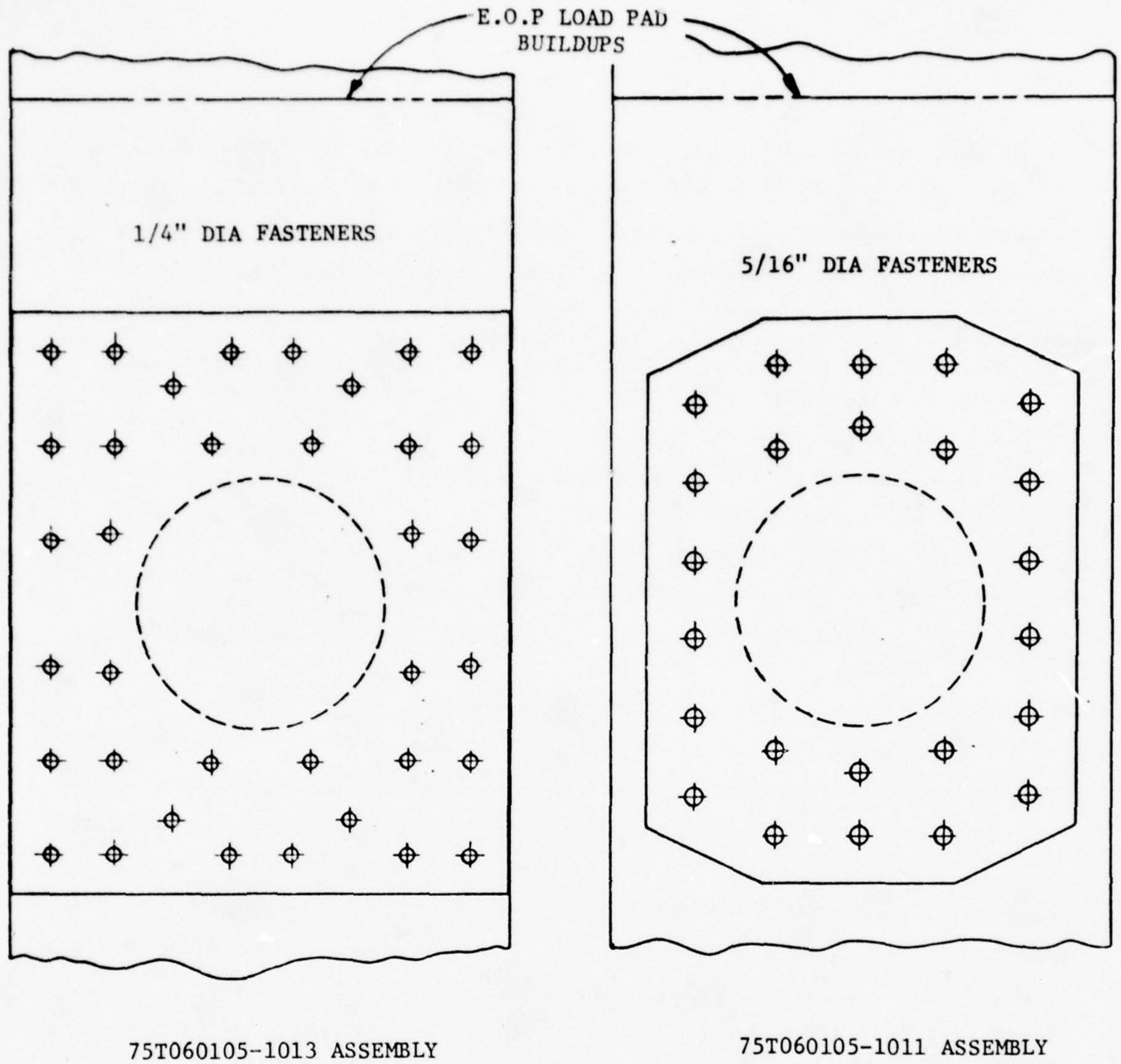
TABLE 3.3 SUMMARY OF TEST DATA - 3/16 LAMINATE - 2.5 DIA. DAMAGE HOLE

3.4 TESTING OF 3/16 INCH LAMINATES WITH 4.0 DIA. DAMAGE HOLE

The two titanium patches designed to repair a 4.0 inch hole in a 3/16 inch laminate are shown in Figure 3-14, along with a table of parts which make up these assemblies. The Engineering drawings (Figures A-1, A-3, and A-5) are found in Appendix A. In designing a patch to repair such a large damage hole, an effort was made to keep the total patch area as compact as possible. This allows application of the repair patch over a larger portion of the wing skin without interfering with adjacent substructure. The two designs presented feature two different sizes of fasteners in similar closely spaced bolt patterns.

The initial test on the 5/16 inch fastener configuration brought failure at 62,500 lb and a gross laminate strain of 4,540 μin./in. The 1/4 in. fastener concept failed at 65,500 lb and 4,758 μin./in. strain. Both failures were along an outermost row of fasteners. Figures 3-15 and 3-16 show the failures. Both concepts thus exceeded 4,000 μin./in., with the 1/4 in. dia. fastener concept slightly better.

It was hoped that further improvement could be made if the number of fasteners in the outermost row were reduced, to reduce the perforated effect along the failure location in the first test. Thus, the 75T060105-1015 assembly was created. This concept is basically the same as the previous 1/4 in. fastener concept (75T060105-1013), except that it had four fasteners



	Drawing Part Number	
	36 Fastener Config (1/4 in. fasteners)	24 Fastener Config (5/16 in. fasteners)
Repair Assy	75T060105-1013	75T060105-1011
Patch Details	75T060103-2005/1005	75T060105-2007/-1007/-1009
Gr/Ep Specimen	75T060101-1005	75T060101-1005

FIGURE 3-14 3/16 LAMINATE 4.0 DIA DAMAGE HOLE INITIAL DESIGNS

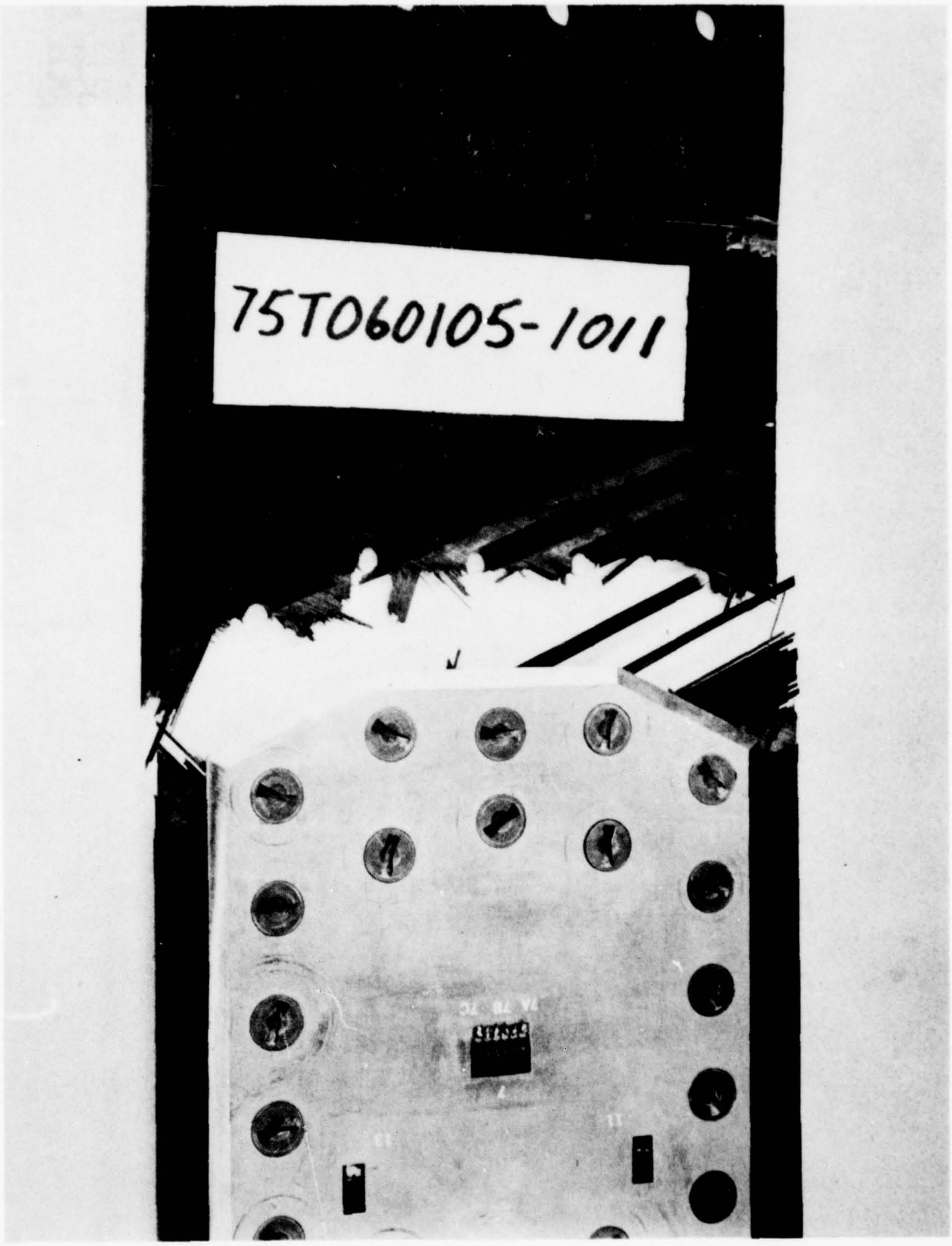


FIGURE 3-15
FAILED 75T060105-1011 SPECIMEN
3/16 Laminate 4.0 Diameter Damage Hole

GP78-8701-10

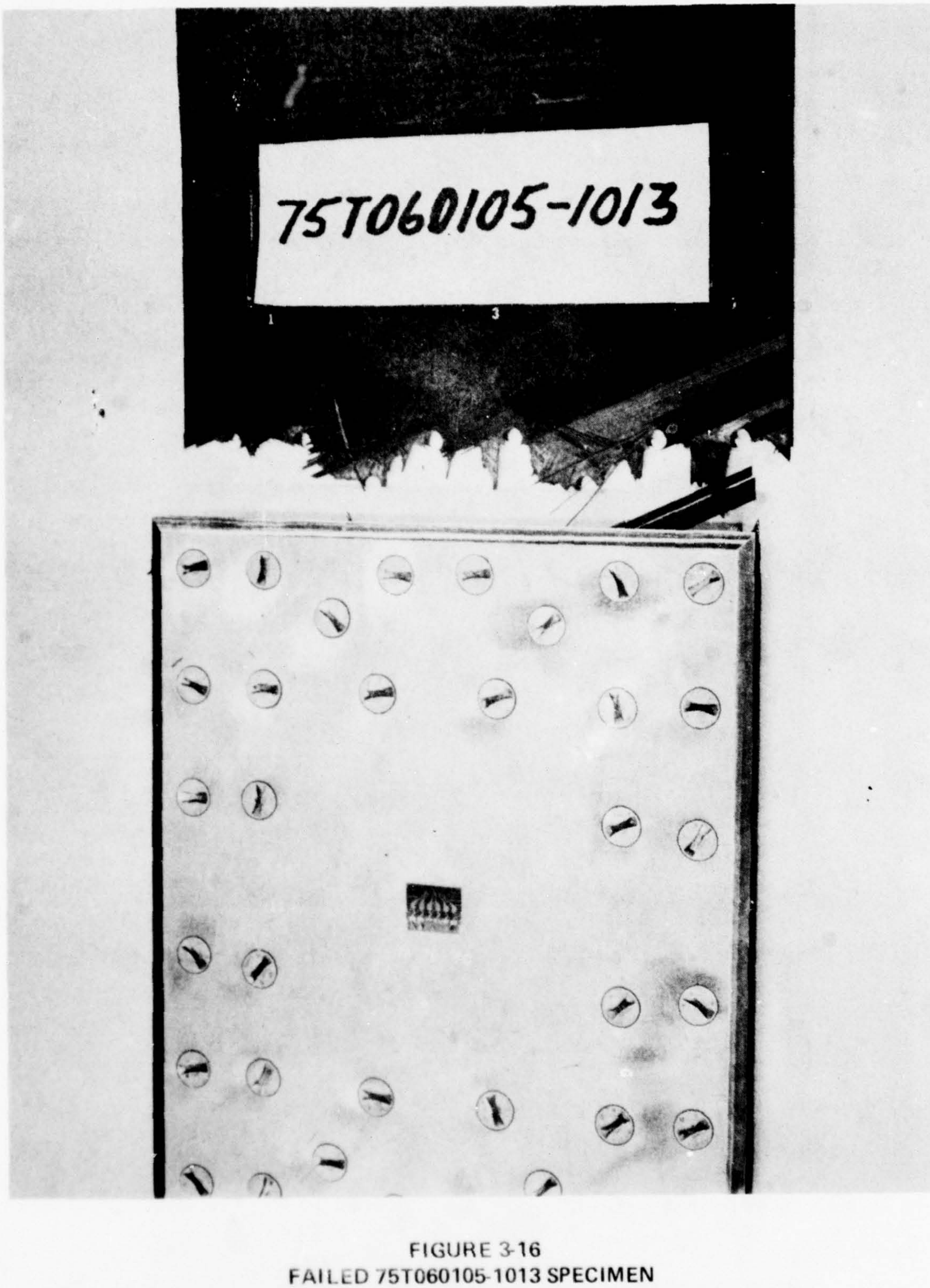


FIGURE 3-16
FAILED 75T060105-1013 SPECIMEN
3/16 Laminate 4.0 Diameter Damage Hole

GP78-8701-11

along each edge instead of six. Figure 3-17 lists the parts in the new assembly. (Reference Figures A-1, A-3, and A-5 in Appendix A), shows the revised fastener pattern.

The follow-on tests showed the revised fastener pattern offered no significant load carrying improvement. Failures occurred at 64,500 lb and 67,500 lb (4758 μin./in. respectively). The failure mode was again along the outer row of fasteners at one end of the patch. This is shown in Figure 3-18. An overall improvement in efficiency can be noted, since the number of fasteners was reduced with no effect on the load-carrying capability of the repair.

Table 3.4 summarizes the most significant data for all the tests. Strain gage data for all tests can be found in Appendix B, Figures B-9 through B-12.

Specimen 75T060105	Failure Load		Strain at Failure, μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1011	62,500	7812	4540	11,389	1376	.30
-1013	65,500	8187	4758	10,220	1700	.36
-1015 #1	64,500	8062	4774	10,731	1546	.32
-1015 #2	67,500	8437	4896	10,208	1565	.32

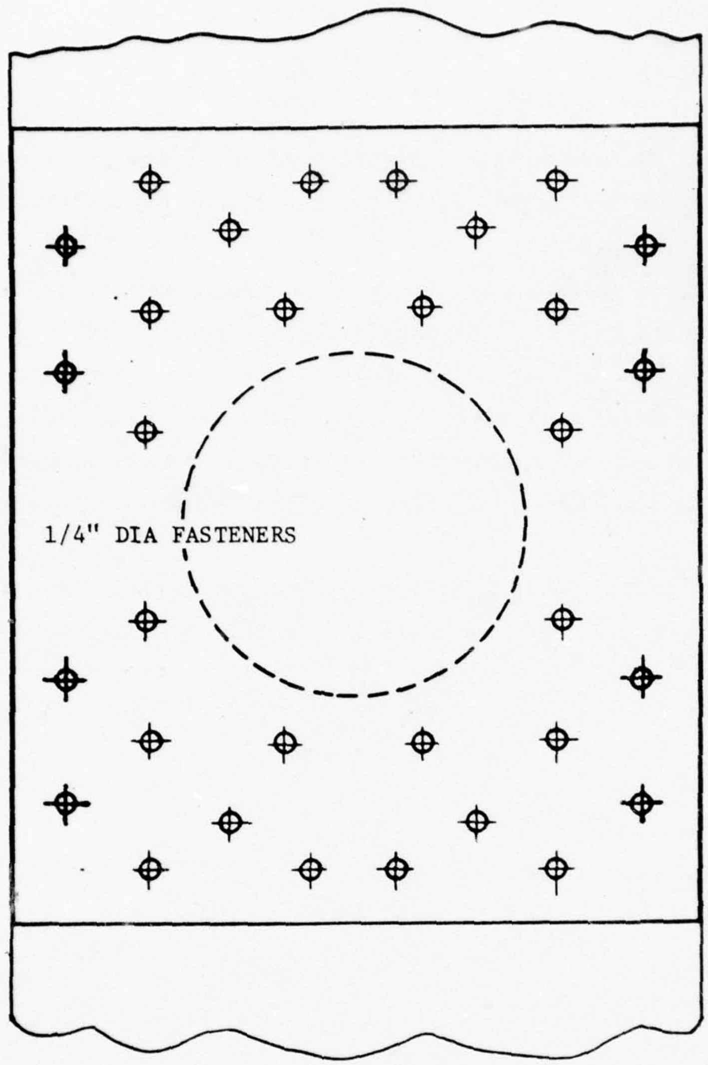
TABLE 3.4 SUMMARY OF TEST DATA - 3/16 LAMINATE 4.0 DIA DAMAGE HOLE

3.5 TESTING OF 3/16 INCH LAMINATE OFF-AXIS SPECIMEN

The repair patch used for the off-axis specimen was chosen based on the initial tests of 3/16 in. laminates with a 2.50 dia damage hole. The best patch was the one with twenty - 1/4 in. fasteners.

The Gr/Ep panel was fabricated with the 0° laminate axis 20° off the load axis. The patch was attached in line with the load. Figure 3-19 shows the assembly and a table of parts. Engineering drawings are found in Figures A-1, A-3 and A-5, Appendix A.

The specimen failed at a load of 73,300 lb and an average laminate strain, in the load direction, of 6,233 μin./in. The specimen failed across the outer row of fasteners. The maximum strain measured was 10,981 μin./in., see Figure 3-20. This test showed the structural adequacy of an off-axis repair. The patch performs similar to its on-axis counterpart (Section 3.3).



75T060105-1015 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060105-1015
Patch Details	75T060103-2013/-1015
Gr/Ep Specimen	75T060101-1005

FIGURE 3-17 3/16 LAMINATE 4.0 DIA DAMAGE HOLE FOLLOW-ON TEST DESIGN

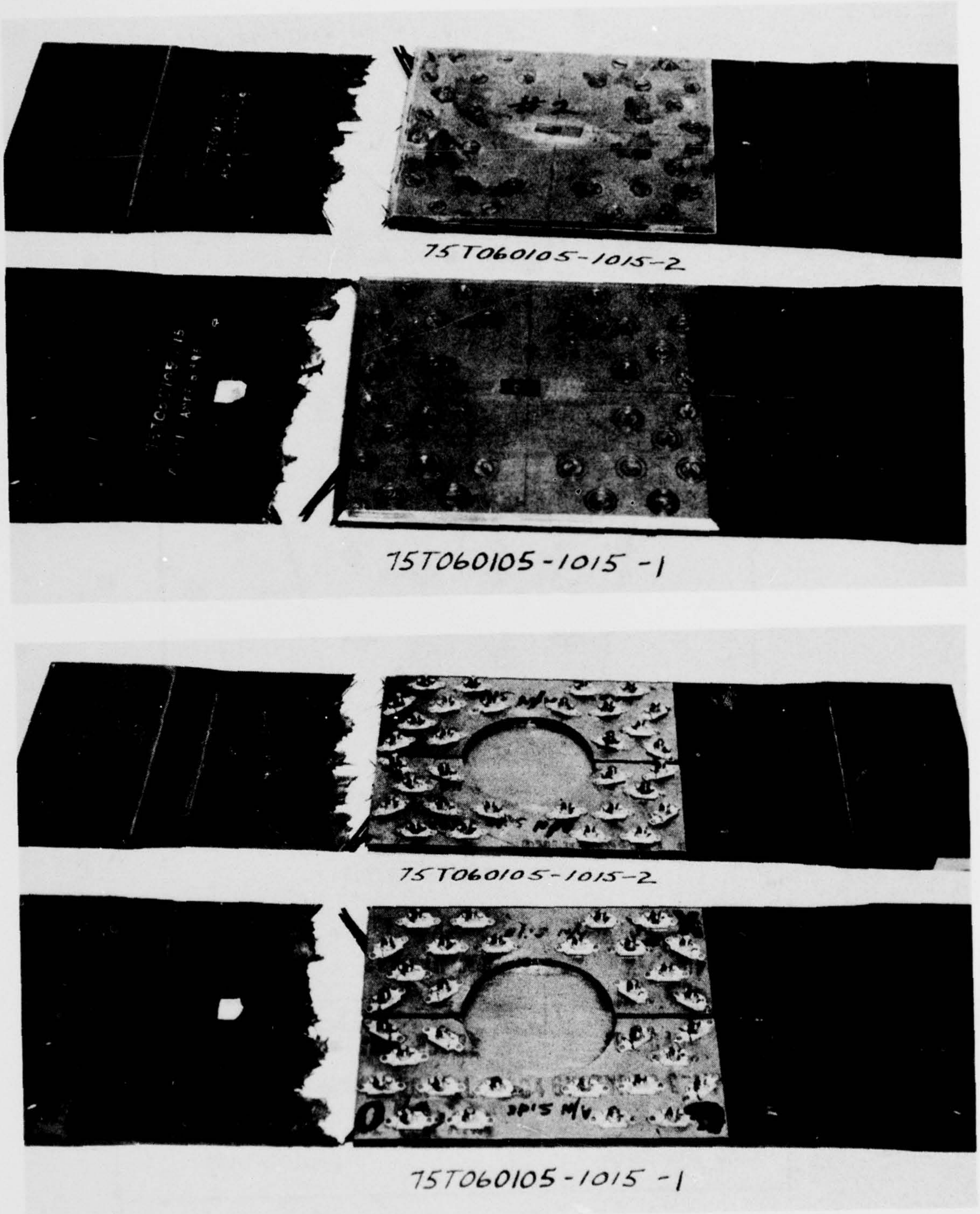


FIGURE 3-18
FAILED 75T060105-1015 SPECIMENS
3/16 Laminate 4.0 Diameter Damage Hole

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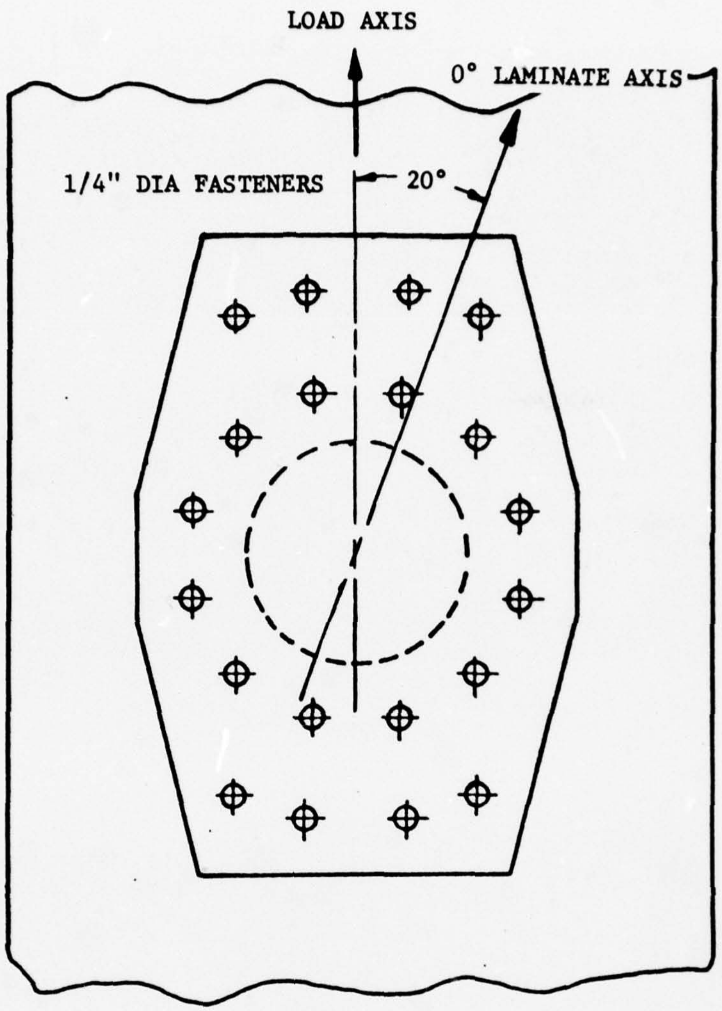

75T060105-1017 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060105-1017
Patch Details	75T060103-2001/-1001
Gr/Ep Specimen	75T060101-1009

FIGURE 3-19 3/16 LAMINATE OFF-AXIS REPAIR DESIGN

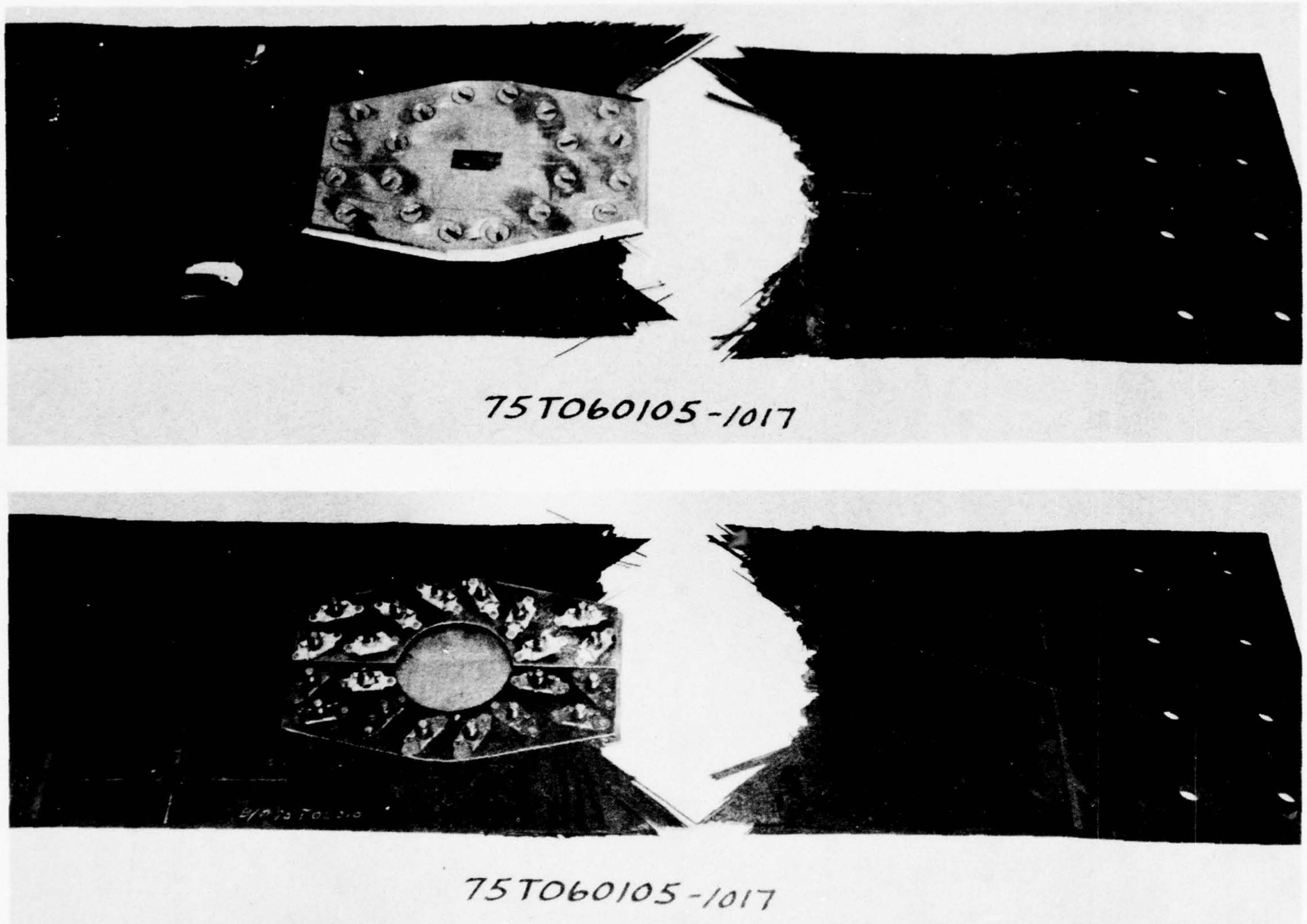

FIGURE 3-20
FAILED 3/16 LAMINATE OFF-AXIS SPECIMEN

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Table 3.5 lists significant test data. (Complete strain gage data is shown in Appendix B, Figure B-13).

Specimen 75T060105	Failure Load		Strain at Failure μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1017	73,700	9,212	6,233	10,981	1,452	.23

TABLE 3.5 SUMMARY OF TEST DATA - 3/16 LAMINATE OFF-AXIS TEST

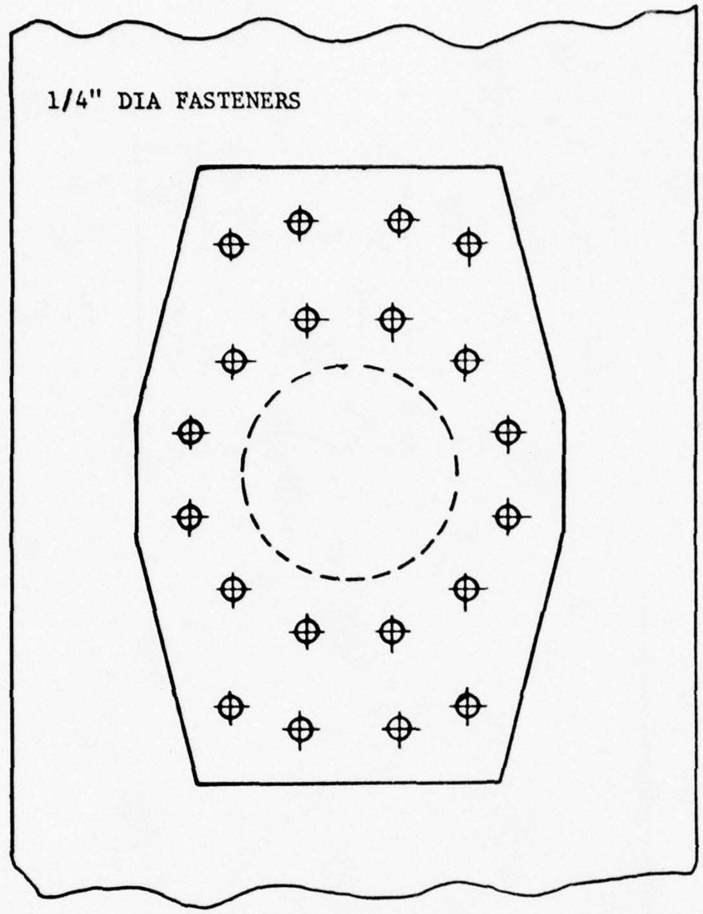
3.6 TESTING OF 3/16 INCH LAMINATE COMPRESSION SPECIMEN

The patch used to repair the hole on the 3/16 in. laminate compression specimen was one with twenty 1/4 in. fasteners used on the tension test of the 2.50 in. hole. Figure 3-21 shows a sketch of the repair and a table of parts.

The specimen was assembled and instrumented according to drawing 75T060105 (Appendix A, Figure A-5). Steel pipe edge supports were clamped in place per drawing 75T060107 (Appendix A, Figure A-7).

The specimen was loaded in compression in 7,000 lb increments. At 35,000 lb examination of the strain gage data showed that the panel was beginning to buckle. The load was held at 35,000 lb and the specimen was examined. Visible buckling had occurred. Along the vertical center line of the specimen the maximum amplitude of the buckling was 0.18 in. Near the edge support, 0.03 in. of the deformation was measured. To preclude damaging the laminate or the patch no further loading was applied. When the load was removed, the specimen returned to its original shape with no discernable permanent deformation.

An examination of the item data from gages 3 and 4, which are back to back along the centerline of the specimen, is shown in Figure 3-22. This shows that buckling began at approximately 21,000 lb load. Table 3.6 summarizes the test data recorded at the highest load level applied.



75T060105-1009 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060105-1009
Patch Details	75T060103-2001/-1001
Gr/Ep Specimen	75T060101-1007

FIGURE 3-21 3/16 LAMINATE COMPRESSION REPAIR DESIGN

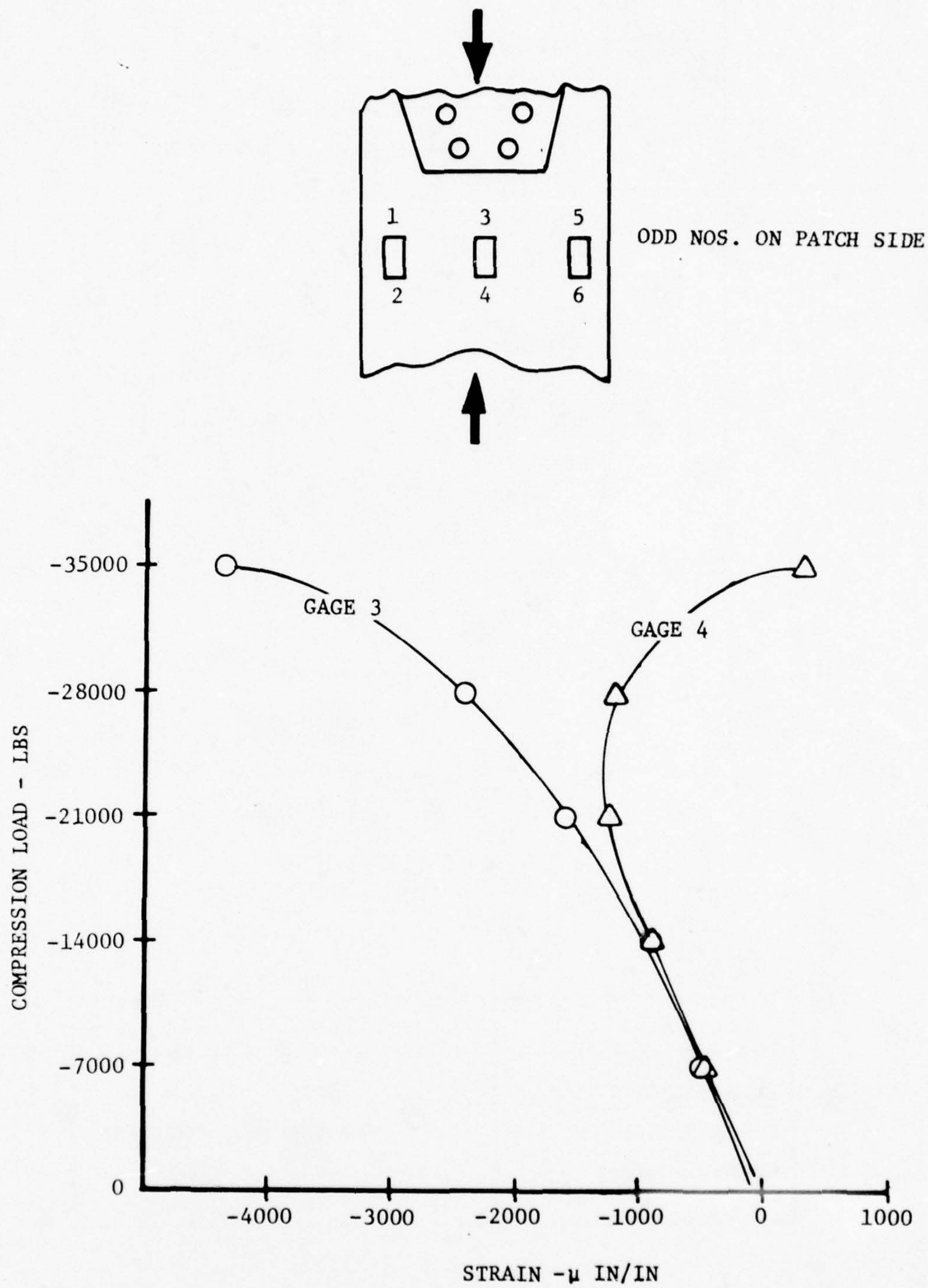


FIGURE 3-22 PARTIAL STRAIN GAGE DATA - 3/16 LAMINATE COMPRESSION TEST

Specimen 75T060105	Max Load Applied		Max. Strains μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate Ave	Hole	Patch	
-1009	-35,000	-4757	-2729	-5450	-385	.14

TABLE 3.6 SUMMARY OF TEST DATA - 3/16 LAMINATE COMPRESSION TEST

3.7 TESTING OF 1/2 INCH LAMINATES WITH 1.0 DIA DAMAGE HOLE

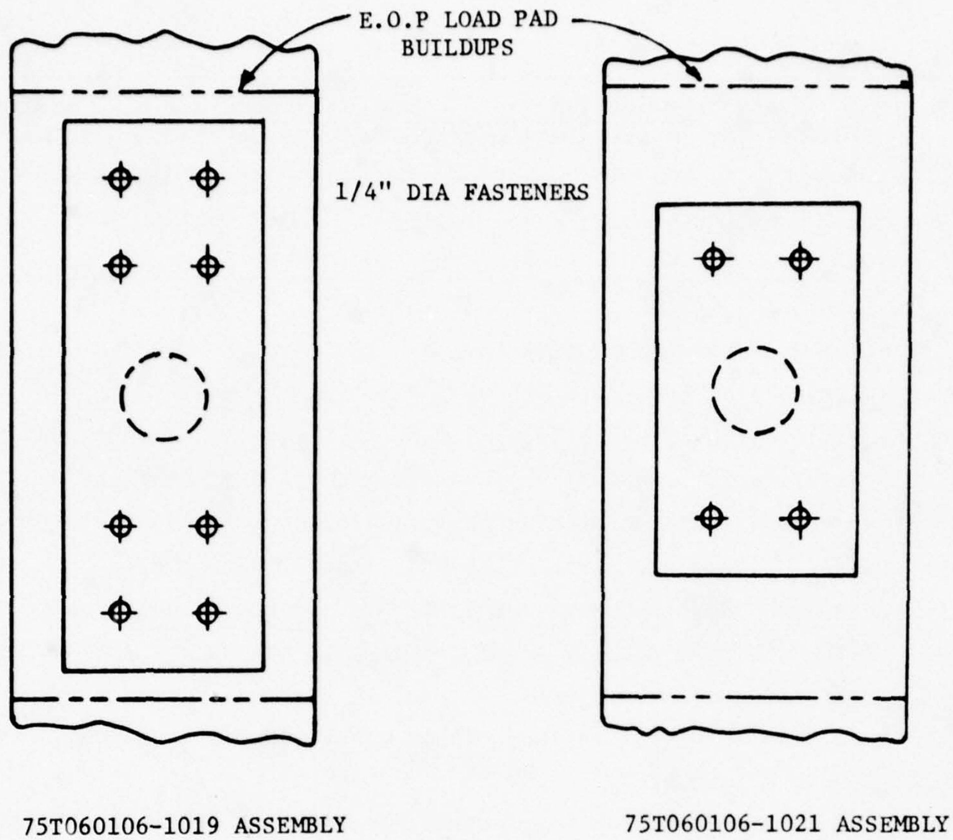
The initial repair patch designs for the 1/2 inch laminate with a 1.0 dia damage hole were identical to the first set of patches tested on the 3/16 laminates, except that the backing plates were thick (0.140 in.) structural items instead of thin nut plate retainers. (See Figure 3-6).

The test of these patches on the 3/16 in. laminates had resulted in a review of the analysis methods and a redesign of the repair patches for all the 1.0 dia damage hole specimens.

Figure 3-23 shows the revised repair patch designs and their parts list for the 1/2 inch laminate.

In the initial tests of the revised patches on a 1/2 inch laminate, the 75T060106-1019 (8 bolt) specimen failed at 97,300 lb and a strain of 5,806 μin./in. while the 75T060106-1021 (4 bolt) specimen went to 98,600 lb and a strain of 6,306 μin./in. The 4-bolt configuration failed across the bolts at one end of the specimen. The 8-bolt design failed across the damage hole. Figure 3-24 shows the failed specimens. One possible reason the 8-bolt patch was not more effective than the 4-bolt configuration is that the length of the patch puts the outer fasteners very near the load pad buildups on the specimen. A complete evaluation of the 8-bolt repair will require further testing on a larger specimen.

Two more 75T060106-1021 (4 bolt) specimens were built and tested. Failures occurred at 94,300 lb and 92,800 lb, with average laminate strains of 5814 and 5832 μin./in. respectively. Table 3.7 summarizes several data parameters from these tests. (Complete strain gage data is in Appendix B, Figures B-15 to B-18).



Part Name	Drawing Part Numbers 8-Bolt Configuration	4-Bolt Configuration
Repair Assy	75T060106-1019	75T060106-1021
Patch Details	75T060104-2011/-1015	75T060104-2013/1017
Gr/Ep Specimen	75T060101-1011	75T060101-1011

FIGURE 3-23 1/2 LAMINATE 1.0 DIA DAMAGE HOLE REVISED PATCH DESIGNS

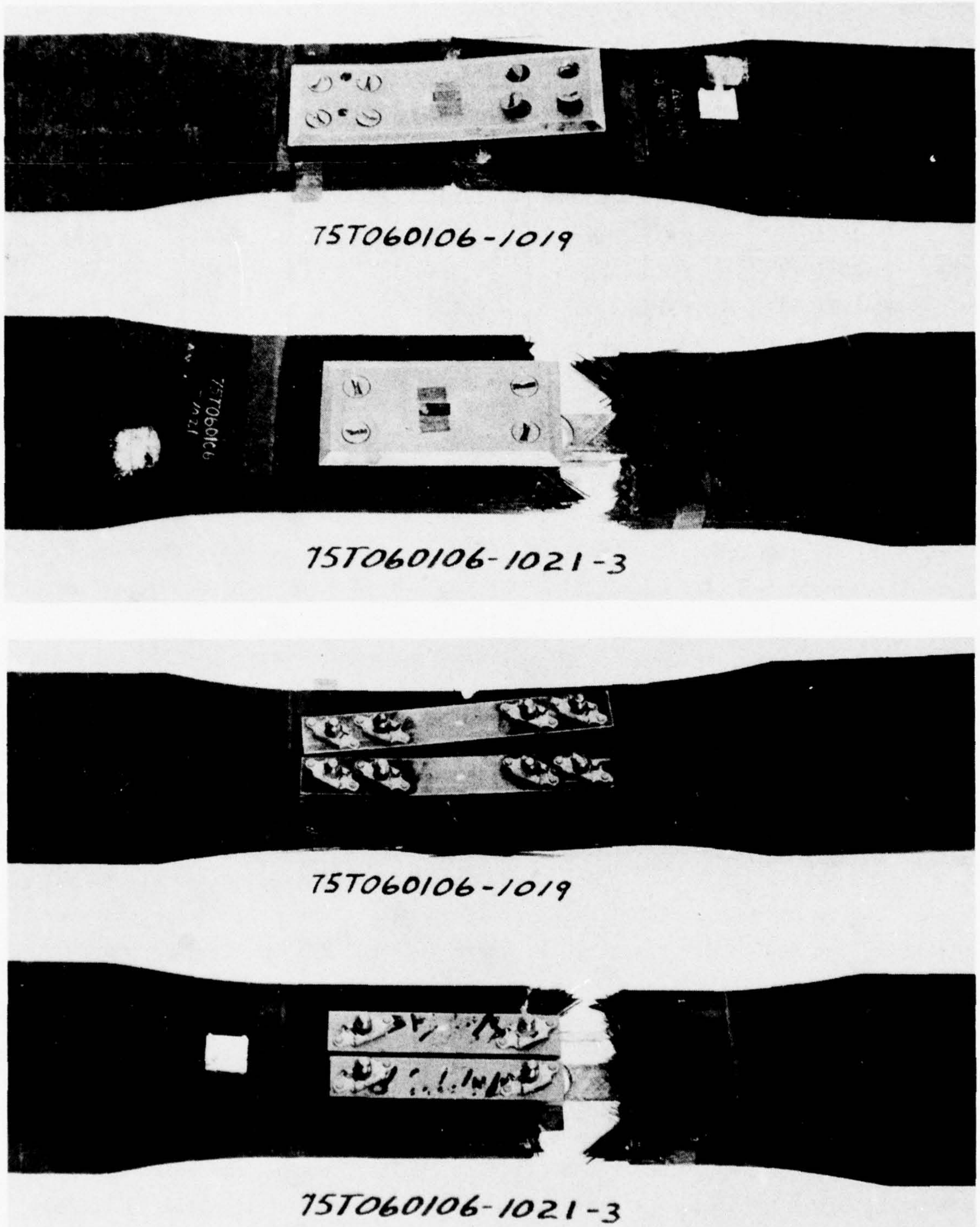


FIGURE 3-24
FAILED SPECIMENS - 1/2 LAMINATE 1.0 DIAMETER DAMAGE HOLE
Initial Tests

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Specimen 75T060106	Failure Load		Strain of Failure μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1019	97,300	27,800	5,805	16,493	1,343	.23
-1021 #1	98,600	28,171	6,306	17,745	827	.13
-1021 #2	94,300	26,942	5,814	-	856	.15
-1021 #3	92,800	26,514	5,832	14,925	903	.15

TABLE 3.7 SUMMARY OF TEST DATA - 1/2 LAMINATE 1.0 DIA DAMAGE HOLE

3.8 TESTING 1/2 INCH LAMINATES WITH 2.50 DIA DAMAGE HOLE

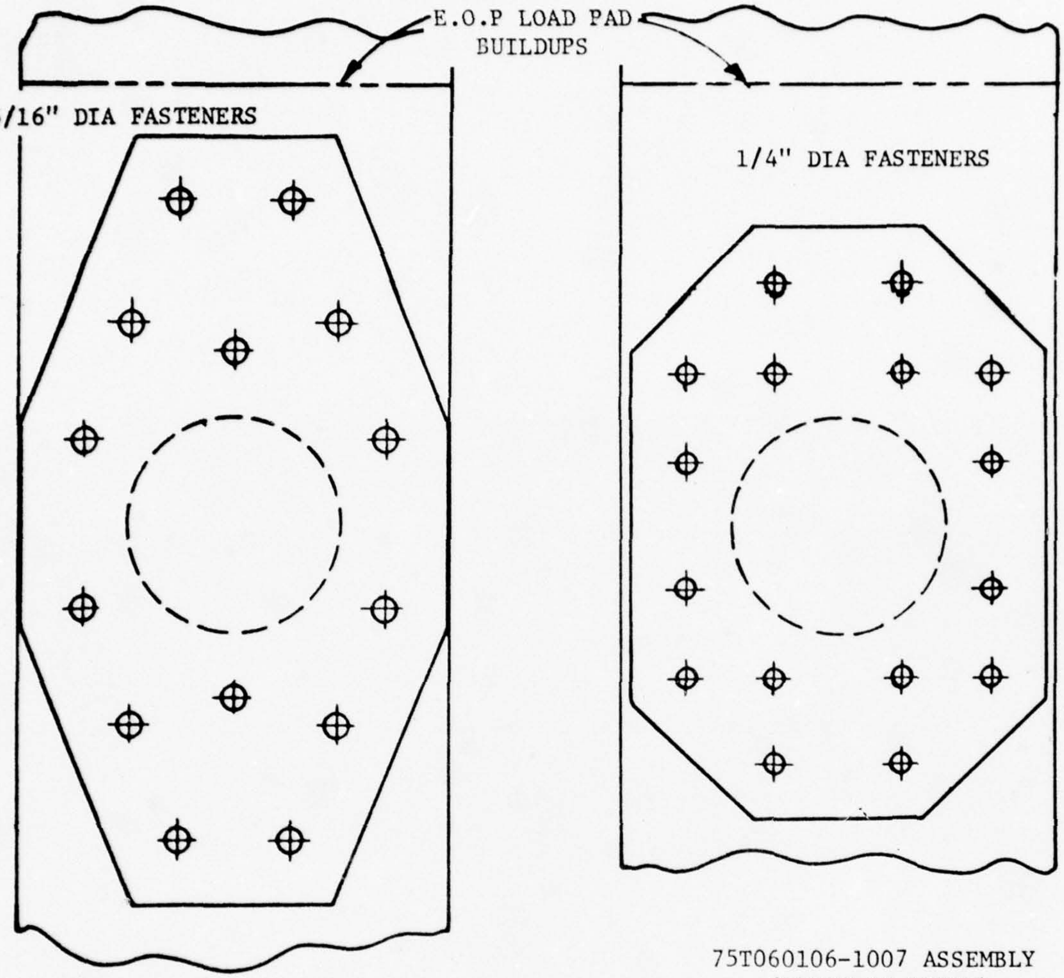
Figure 3-25 shows the two repair assemblies for a 1/2 in. laminate with a 2.50 in. hole. The table included in that figure lists the components in the assemblies (Ref Appendix A, Figures A-1, A-4 and A-6) developed using 1/4 in. dia bolts in one patch and 5/16 in. dia bolts in the other.

Initial tests of these specimens show the 75T060106-1005 (5/16 inch dia fastener) configuration failed at 91,750 lb and 3991 μin./in. average laminate strain. Failure was in the net section across the damage hole, See Figure 3-26. The 75T060106-1007 specimen (1/4 in. fasteners) failed at 91,500 lb and 3829 μin./in. strain. The failure mode also occurred in the net section of the hole. See Figure 3-27.

The failed loads and laminate strains for these two configurations were essentially the same. The 1/4 in. fastener repair was chosen for follow-on testing because in all prior tests, where 5/16 and 1/4 in. fasteners were compared, the smaller fastener had produced better results.

The configuration was also chosen to test a stiffer patch material. It was hoped that a stiffer patch would pick up more of the load, thus relieving the laminate in the damage area.

The first follow-on test was performed on a repair which was identical in geometry to the 75T060106-1007 (1/4 in. bolts), except the patch and backing plate were made of stainless steel. The modulus of steel is roughly twice that of titanium, and is the same material used for repair patches on F-4 aircraft. The assembly is the 75T060106-1017 specimen as noted in Figure 3-25.



75T060106-1007 ASSEMBLY
(TITANIUM PATCH)

75T060106-1017 ASSEMBLY*
(STEEL PATCH)

*ALTERNATE CONFIGURATION TESTED DURING
FOLLOW-ON TEST PHASE (1 SPECIMEN)

Part Name	Drawing Part Numbers 5/16 in. Fastener Config	 1/4 in. Fastener Config
Repair Assy	75T060106-1005	75T060106-1007
Patch Details	75T060104-2003/-1007	75T060104-2001/-1005
Gr/Ep Specimen	75T060101-1013	75T060101-1013

FIGURE 3-25 1/2 LAMINATE 2.5 DIA DAMAGE HOLE REPAIR DESIGNS

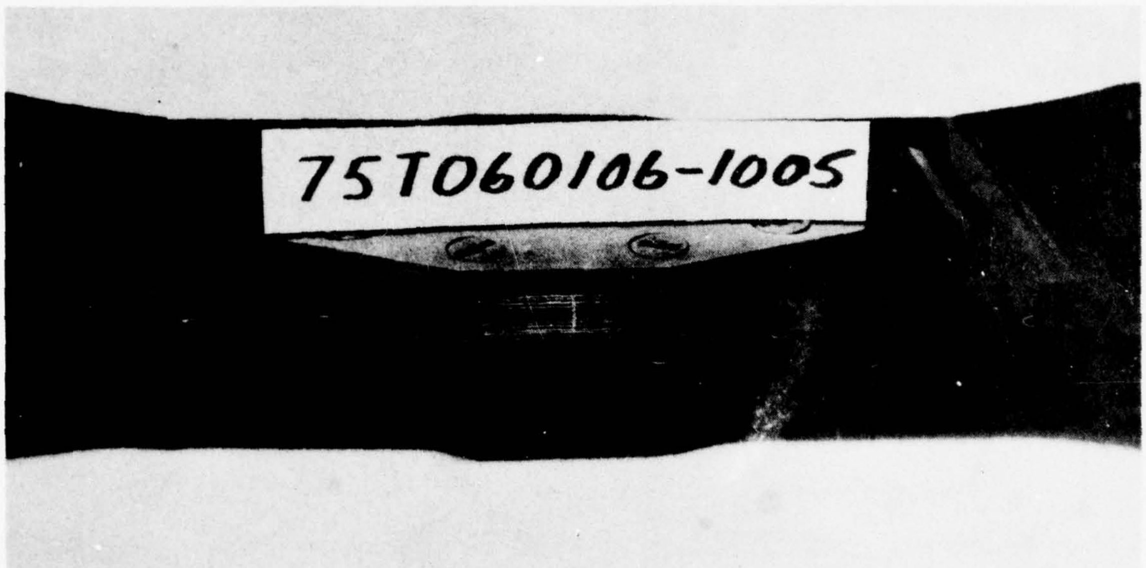


FIGURE 3-26
FAILED 75T060106-1005 SPECIMEN
1/2 Laminate 2.5 Diameter Damage Hole

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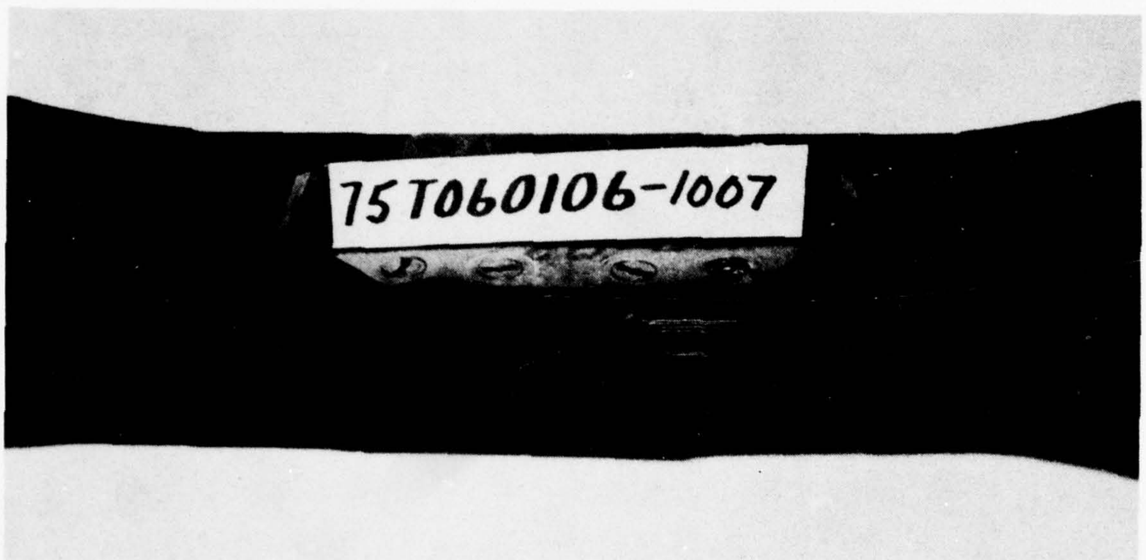


FIGURE 3-27
FAILED 75T060106-1007 SPECIMEN
1/2 Laminate 2.5 Diameter Damage Hole

GP78-8701-13

Testing the assembly with the steel patch revealed no significant improvement. The specimen failed at 92,750 lb, with an average laminate strain of 3828 μin./in. Measured patch strain was 909 μin./in. which, when ratioed according to modulus, compares exactly to the strain in the titanium patch. The specimen failed in the net section across the damage hole.

Since no improvement was noted, the final specimen was tested with a titanium patch. Table 3.8 summarizes the results: (Complete data is in Appendix B, Figures B-19 to B-22)

Specimen 75T060106	Failure Load		Strain of Failure μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1005	91750	18350	3991	12060	1544	.39
-1007	91500	18350	3829	13168	1647	.43
-1017	92750	18550	3828	13522	909/1648*	.24 / .43*
-1007 #2	93100	18187	3844	13748	1520	.40

*(Ratioed up by 29.0/16.0 - modulus of steel/modulus of titanium)

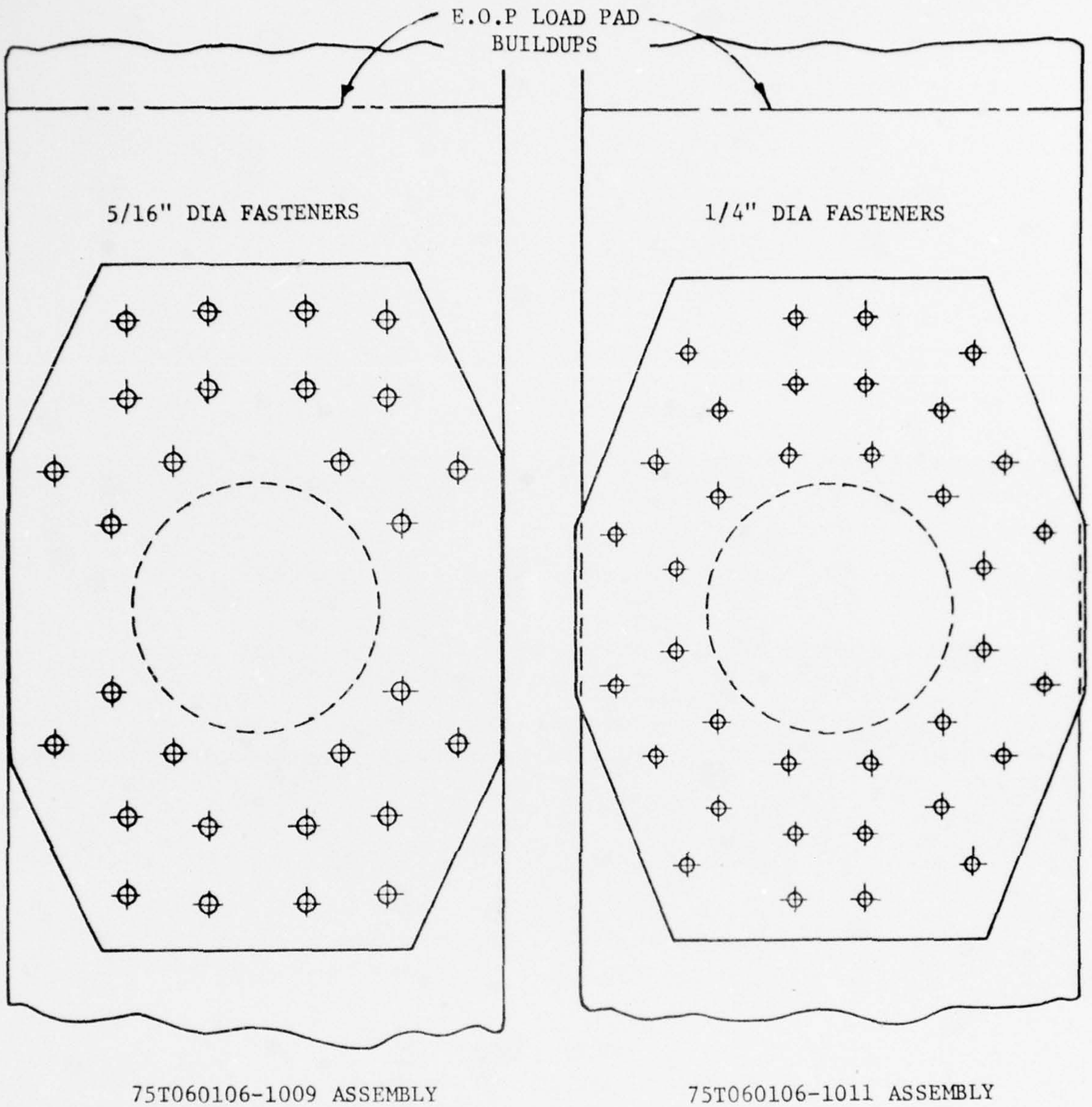
TABLE 3.8 SUMMARY OF TEST DATA - 1/2 LAMINATE - 2.5 DIA DAMAGE HOLE

3.9 TESTING OF 1/2 INCH LAMINATES WITH 4.0 DIA DAMAGE HOLE

The two repair configurations designed for a 1/2 in. laminate with a 4.0 inch dia damage hole are sketched in Figure 3-28. The table in that figure list the detailed parts which go into each assembly (Ref Appendix A, Figures A-1, A-4 and A-6). One configuration, 75T060106-1009 has twenty-eight 5/16 in. fasteners. The other repair patch has thirty-six 1/4 in. fasteners (75T060106-1011).

In the intial test phase the 75T060106-1009 (5/16 fastener) repair failed at 145,500 lb and an average laminate strain of 3876 μin./in. The laminate failed across the net section of the damage hole area as shown in Figure 3-29. The strain in the patch at failure was 1920 μin./in.

The 75T060106-1011 repair failed at 156,500 lb and 4050 μin./in. average laminate strain. Failure in the panel was also across the damage hole net section as shown in Figure 3-30. At failure most of the patch fasteners on half of the patch deformed severely and five bolt heads broke off.



Part Name	Drawing Part Numbers 5/16 in. Fastener Config (28 Fasteners)	 1/4 in. Fastener Config (36 Fasteners)
Repair Assy	75T060106-1009	75T060106-1011
Patch Details	75T060104-2005/-1009	75T060104-2007/-1011
Gr/Ep Specimen	75T060101-1015	75T060101-1015

FIGURE 3-28 1/2 LAMINATE 4.0 DIA DAMAGE HOLE REPAIR DESIGNS

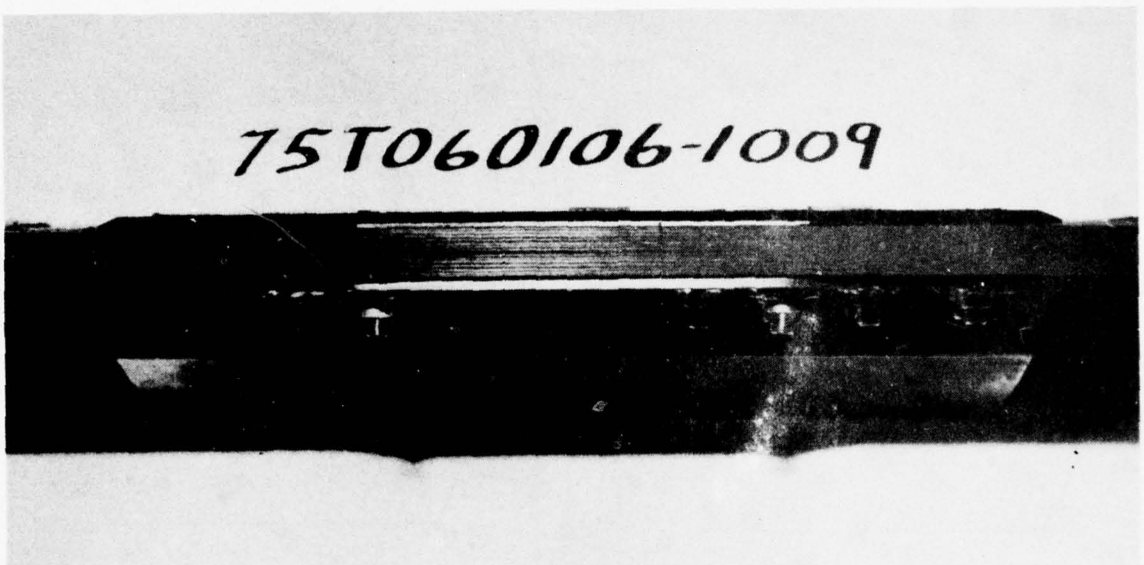


FIGURE 3-29
FAILED 75T060106-1009 SPECIMEN
1/2 Laminate 4.0 Diameter Damage Hole

GP78-8701-14

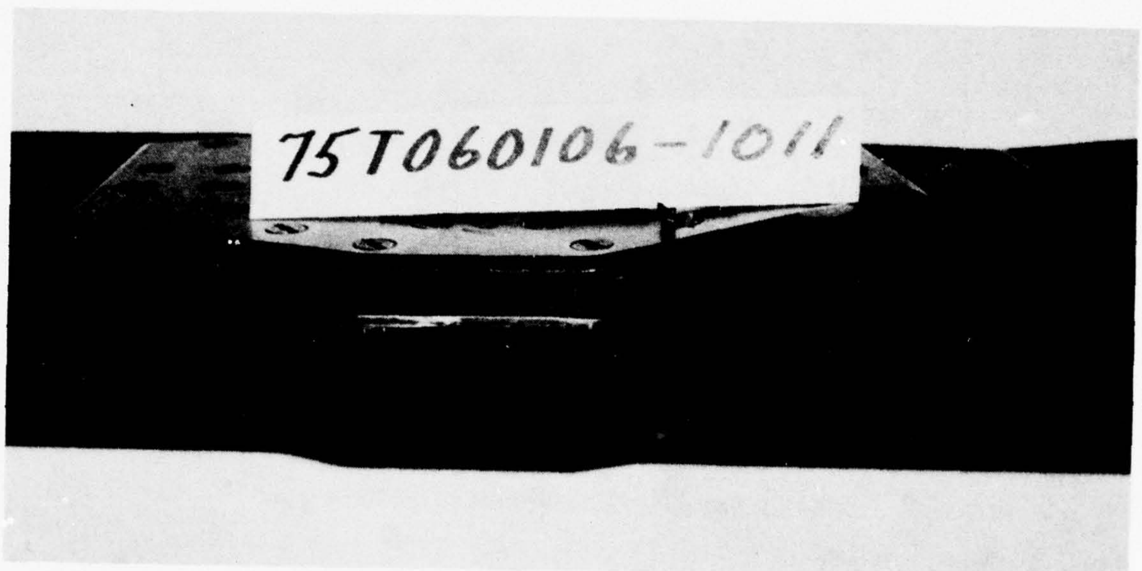


FIGURE 3-30
FAILED 75T060106-1011 SPECIMEN
1/2 Laminate 4.0 Diameter Damage Hole

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The strain in the patch at failure was 2868 μin./in.

The 75T060106-1011 (1/4 in. fastener) repair withstood a higher load and the patch strain was considerably higher, so it was chosen for follow-on testing. The follow-on testing produced failures at 152,750 and 167,750 lb with strains in the laminate of 3995 and 4325 μin./in. respectively. Failure modes were identical to the first specimen of that configuration. Table 3.9 summarizes the significant test data. (Complete strain data is in Appendix B, Figures B-23 to B-26).

Specimen 75T060106	Failure Load		Strain of Failure μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1009	145,500	18,187	3872	9,579	1920	.50
-1011 #1	156,500	19,560	4050	12,192	2868	.71
-1011 #2	152,750	19,093	3995	11,500	2459	.62
-1011 #3	167,750	20,969	4325	13,323	2692	.62

TABLE 3.9 SUMMARY OF TEST DATA - 1/2 LAMINATE - 4.0 DIA DAMAGE HOLE

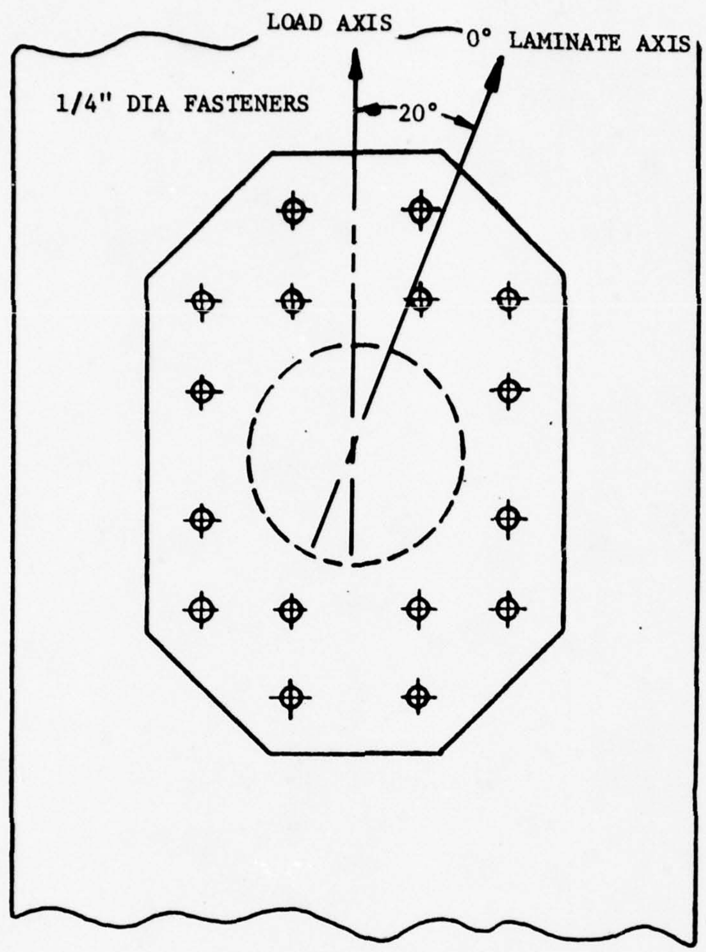
3.10 TESTING OF 1/2 INCH LAMINATE OFF-AXIS SPECIMEN

The patch chosen for the 1/2 in. laminate off-axis specimen was the 1/4 in. fastener design for a 2.50 in. hole. Figure 3-31 is a drawing of the patch configuration and a table listing the parts in the assembly. (Refer to the engineering drawings, Figures A-1, A-4, and A-6 in Appendix A.

The laminate was fabricated with the 0° axis of the layup oriented 20° off the load direction. The patch was mounted in line with the load line.

The specimen failed at 150,000 lb and a strain of 4,178 μin./in. The patch strain was 1,502 μin./in. at failure while the strain at the edge of the hole was 15,078 μin./in. The laminate failed through the damage hole along a line roughly perpendicular to the 0° axis of the laminate. Figure 3-32 is a photograph of the failed specimen.

Table 3.10 summarizes the test results. Complete strain gage data is presented in Appendix B, Figure B-27.



75T060106-1013 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060106-1013
Patch Details	75T060104-2001/-1005
Gr/Ep Specimen	75T060101-1019

FIGURE 3-31 1/2 LAMINATE OFF-AXIS REPAIR DESIGN

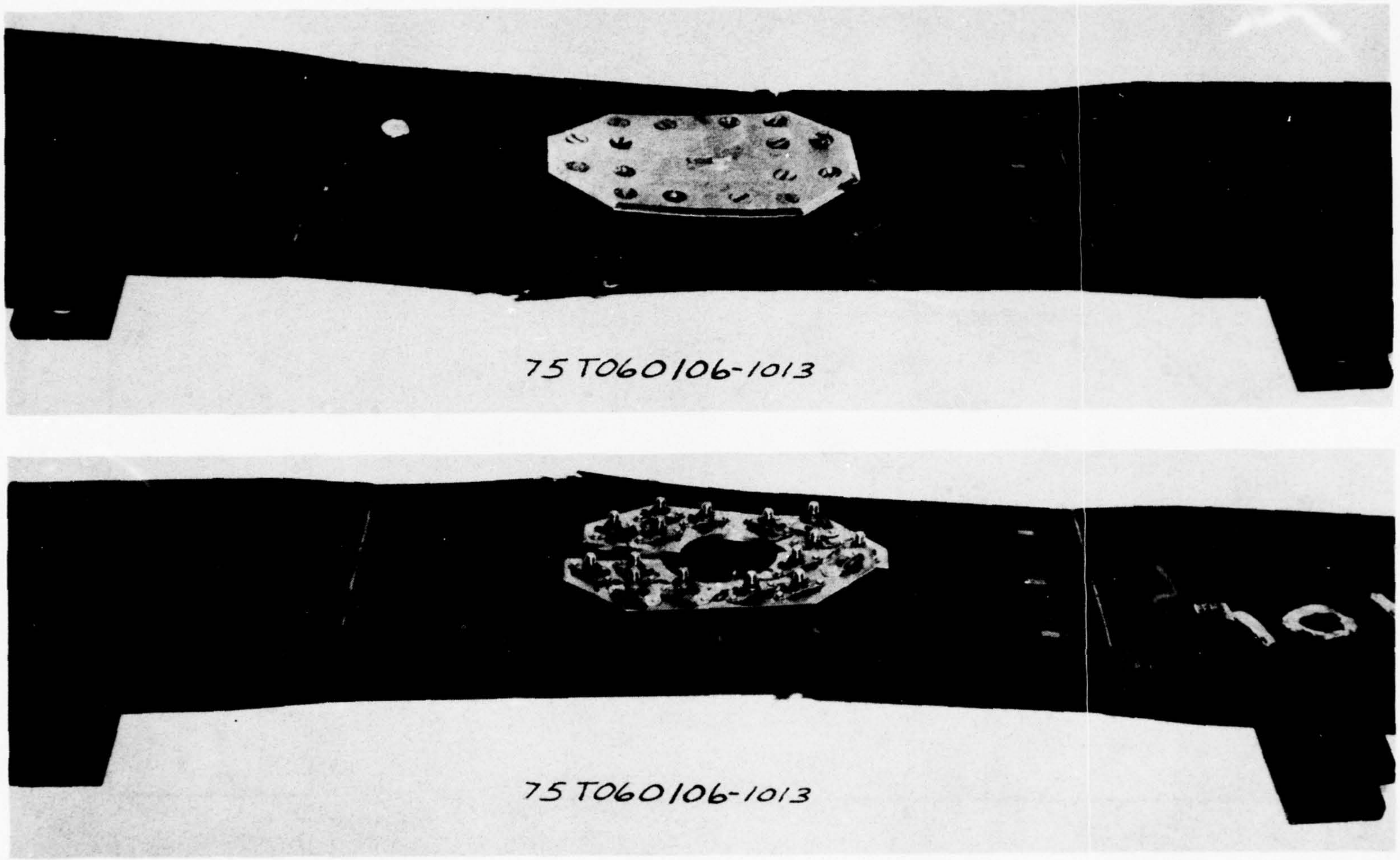


FIGURE 3-32
FAILED 1/2 LAMINATE OFF-AXIS SPECIMEN

GP78 8701-17

Specimen 75T060106	Failure Load		Strain at Failure μin./in.			Stress Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1013	150,000	18,750	4,178	15,078	1,502	.36

TABLE 3.10 SUMMARY OF TEST DATA - 1/2 LAMINATE OFF-AXIS TEST

3.11 TESTING OF 1/2 INCH LAMINATE COMPRESSION SPECIMEN

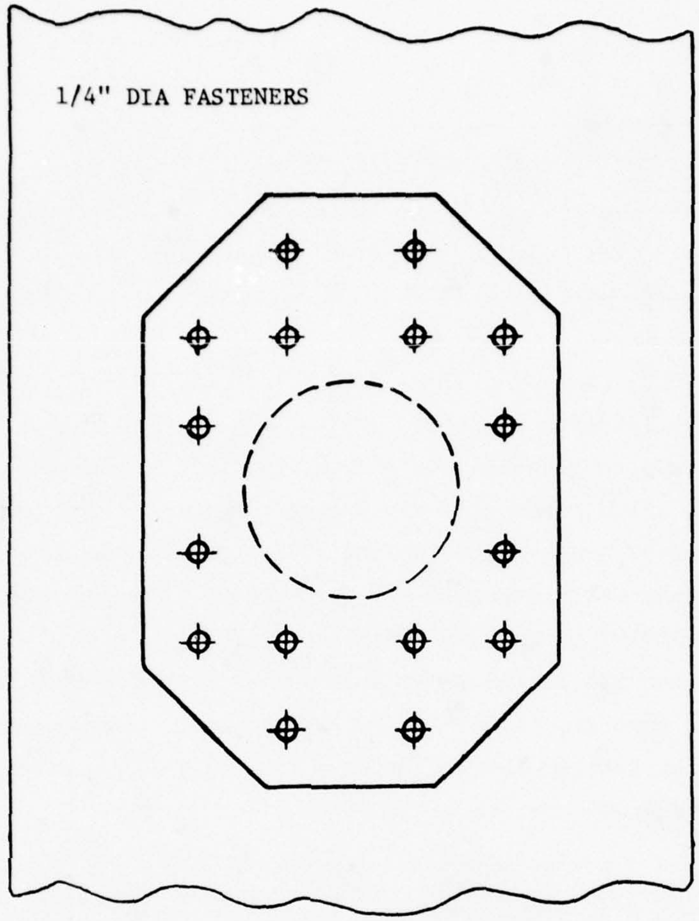
The repair patch used on the 1/2 in. laminate compression specimen was the design which performed best on the tension test of the 2.50 in. dia hole. Figure 3-33 shows the patch and a list of component parts.

The specimen was restrained along its edges by support pipes described in drawing 75T060107 (Figure A-7, Appendix A).

The specimen was loaded to an average laminate strain of 4,000 μin./in. while data was recorded, then the load was removed. The undamaged specimen is scheduled for fatigue testing later. Table 3.11 summarizes the data recorded at the 4,000 μin./in. load level.

Specimen 75T060106	Max. Load Applied		Max. Strain μin./in.			Strain Ratio Patch/Laminate
	lb	lb/in.	Laminate	Hole	Patch	
-1015	154,000	19250	4042	11626	1621	.40

TABLE 3.11 SUMMARY OF TEST DATA - 1/2 LAMINATE COMPRESSION TEST



75T060106-1015 ASSEMBLY

Part Name	Drawing Part Numbers
Repair Assy	75T060106-1015
Patch Details	75T060104-2001/-1005
Gr/Ep Specimen	75T060101-1017

FIGURE 3-33 1/2 LAMINATE COMPRESSION REPAIR DESIGN

4. EVALUATION OF RESULTS

Table 4-1 summarizes the significant test data. The 3/16 in. laminates, except for one specimen, all demonstrated a laminate strain level in excess of the target 4,000 μin./in. strain. Eight out of fourteen 1/2 in. thick repaired laminates exceeded the 4,000 μin./in. target strain level and the remaining 6 specimens all reached within 4% of the target strain.

4.1 EFFECTIVENESS OF REPAIRS

The effectiveness of a laminate repair can be obtained by dividing the gross strain of the repaired laminates at failure by the average failing strain of similar undamaged laminates. In this study, the data for an undamaged laminate were taken from the coupon tests peformed for the YAV-8B/AV-8B program Ref (1), which showed an average failing strain of 5200 μin./in. This value was used to calculate a repair effectiveness value for each specimen tested which is shown in Table 4.2.

It can be seen that the repairs restored the laminates to between 74% and 121% of the baseline strength of an undamaged panel. In actuality, of course, none of the laminates were restored to more than 100% baseline strength. Any values of repair effectiveness beyond unity can be attributed to normal material variability which, due to the limited number of specimens of each concept tested, was not accounted for in this study.

Evaluating the performance of the repair concepts solely on the basis of the gross failure strain may be misleading because the degree of improvement required for the repair varies with the size of the damage. For instance, as shown in Table 2.2, the hole edge failure strain of an unrepaired panel containing a 2.5 in. hole is 42% lower than that of a similar panel with a 1.0 in. diameter hole. Consequently, the repair for the panel with the 1.0 in. hole need not be as effective as that for a 2.5 in. hole in order to achieve the same gross strain level.

An improvement ratio was calculated for each specimen by dividing the actual average laminate strain at failure by the calculated maximum strain of an unrepaired laminate. Table 4.2 summarizes these data.

3/16 LAMINATES TEST DATA

DAMAGE HOLE DIAMETER	SPECIMEN 75T060105	FAILURE LOAD		STRAIN AT FAILURE μin./in.			STRAIN RATIO PATCH/LAMINATE
		LB	LB/IN.	LAMINATE	HOLE	PATCH	
1.0	-1001	23,300	6,714	3,846	19,914	30	0.01
1.0	-1003	25,000	7,142	4,039	+10,000 ⚠1	200	0.05
2.5	-1005#1	46,000	9,220	5,316	+10,000 ⚠1	1,569	0.30
2.5	-1005#2	42,100	8,420	4,893	-	1,378	0.28
2.5	-1005#3	46,600	9,320	5,358	15,441	1,485	0.28
2.5	-1007	36,400	7,280	4,291	13,605	873	0.20
2.5 ⚠4	-1009 ⚠2	(35,000)	(4,375)	-2,720	-5,430	385 ⚠3	0.14
4.0	-1011	62,500	7,812	4,540	11,389	1,376	0.30
4.0	-1013	65,500	8,187	4,758	10,220	1,700	0.36
4.0	-1015#1	64,500	8,062	4,774	10,731	1,546	0.32
4.0	-1015#2	67,500	8,437	4,896	10,208	1,565	0.32
2.5 ⚠5	-1017	73,700	9,212	6,233	10,981	1,452	0.23
1.0	-1019#1	37,000	10,571	5,712	15,175	875	0.15
1.0	-1019#2	35,600	10,171	5,604	13,083	980	0.17

⚠1 Strain Range Set at 10,000 μ in/in.
⚠2 Panel Buckled
⚠3 Patch Buckled
⚠4 Compression Specimen
⚠5 Off Axis Specimen

NOTE: Complete strain gage data is presented in Appendix B

TABLE 4.1 - SUMMARY OF REPAIR TEST DATA

1/2 LAMINATES TEST DATA

DAMAGE HOLE DIAMETER	SPECIMEN 75T060106	FAILURE LOAD		STRAIN AT FAILURE μin./in.			STRAIN RATIO PATCH LAMINATE
		LB	LB/IN.	LAMINATE	HOLE	PATCH	
2.5	-1005	91,750	18,350	3,991	12,060	1,554	0.39
2.5	-1007#1	91,500	18,350	3,829	13,168	1,647	0.43
2.5	-1007#2	93,100	18,620	3,844	13,748	1,520	0.40
4.0	-1009	145,500	18,187	3,872	9,579	1,920	0.50
4.0	-1011#1	156,500	19,560	4,050	12,192	2,868	0.71
4.0	-1011#2	152,750	19,093	3,995	11,500	2,459	0.62
4.0	-1011#3	167,750	20,969	4,325	13,323	2,692	0.62
2.5 △2	-1013	150,000	18,750	4,178	15,078	1,502	0.36
2.5 △3	-1015 △1	154,000	19,250	4,042	11,626	1,621	0.40
2.5 △4	-1017	92,750	18,550	3,828	13,522	909	.24
1.0	-1019	97,300	27,800	5,806	16,493	1,343	.23
1.0	-1021#1	98,600	28,171	6,306	17,745	827	0.13
1.0	-1021#2	94,300	26,942	5,814	(14,879)	856	0.15
1.0	-1021#3	92,800	26,514	5,832	14,925	903	0.15

△1 Specimen Not Tested to Failure
△2 Off Axis Specimen
△3 Compression Specimen
△4 Steel Patch

NOTE: Complete strain gage data is presented in Appendix B

TABLE 4.1 - SUMMARY OF REPAIR TEST DATA (Continued)

SPECIMEN	PANEL THICKNESS (INCH)	HOLE DIAMETER (INCH)	REPAIR EFFECTIVENESS (1)	IMPROVEMENT RATIO (2)
75T060105-1001	3/16	1.0	.77	.97
-1003		1.0	.77	1.04
-1005-1		2.5	1.02	1.93
-1005-2		2.5	.93	1.78
-1005-3		2.5	1.01	1.94
-1009		2.5	.83	1.56
-1011		4.0	.87	1.71
-1013		4.0	.91	1.79
-1015-1		4.0	.92	1.80
-1015-2		4.0	.94	1.85
-1017		2.5	1.20	2.26
-1019-1		1.0	1.10	1.46
-1019-2	3/16	1.0	1.08	1.44
75T060106-1001	1/2	1.0	(3)	(3)
-1003		1.0	(3)	(3)
-1005		2.5	.78	1.54
-1007-1		2.5	.74	1.39
-1007-2		2.5	.74	1.39
-1009		4.0	.75	1.46
-1011-1		4.0	.78	1.53
-1011-2		4.0	.77	1.51
-1011-3		4.0	.83	1.63
-1013		2.5	.80	1.52
-1017		2.5	.74	1.39
-1019		1.0	1.12	1.49
-1021-1		1.0	1.21	1.62
-1021-2		1.0	1.12	1.49
-1021-3	1/2	1.0	1.12	1.50

(1) STRENGTH OF REPAIRED PANEL/STRENGTH OF BASELINE PANEL

(2) STRENGTH OF REPAIRED PANEL/STRENGTH OF UNREPAIRED PANEL

(3) NOT FABRICATED. REPLACED BY 75T060106-1019 AND -1021 SPECIMENS.

TABLE 4.2 SUMMARY OF REPAIR PERFORMANCE

The improvement ratio summary, with the exception of the 75T060105-1001 and -1003 specimens, shows that the repaired specimens were improved by at least 39% over an unrepaired specimen. Furthermore, specimens 75T060106-1005, -1007, -1013 and -1017 for the 2.5 in. hole, while failing to attain the target strain level of 4,000 μin./in., had improvement ratios comparable to the 75T060106-1021 specimens, all of which exceeded a gross strain of 5,800 μin./in. before failing. Consequently, it must be concluded that the repairs for the 1/2 in. laminates with 2.5 in. holes were as efficient as those for the 1.0 in. holes in spite of the large difference in strain levels. Using the improvement ratio in this manner, the relative performance of differing repair designs for different hole diameters can be compared.

4.2 COMPARISON BETWEEN THEORY AND TEST

Table 4.3 summarizes a comparison between theoretical predictions and test results. Shown are the predicted and measured strains at the hole edge when the gross laminate strain is 4,000 μin./in. and the predicted and measured gross strain level at failure. The comparison of the hole strain serves as a measure of the validity of the compliance model analysis procedure. The comparison of the gross failure strains serves as a measure of the validity of the strength criterion applied to the specimen.

Table 4.3 shows that with the exception of the 75T060106-1021-1, -2, and -3 the predicted and measured strains agree to within 5 to 20%. The agreement is considered to be satisfactory in view of the complexity of the analysis and the many simplifications necessary to make it tractable.

It is significant that the analysis showed a high sensitivity to fastener clearance. The drawing tolerances on the fasteners and holes allow clearance from 0.000 to 0.0027 inches. No attempt was made to control or verify these tolerances beyond normal shop procedures.

Frictional clamp-up between the patch and panel was ignored. Careful analysis of the test data shows evidence that frictional forces did exist. Figure 4-1 shows a plot of load vs. the ratio of patch strain to laminate strain for the 75T060105-1003 specimen. In the absence of friction and fastener clearance, the strain ratio would be constant for all load levels. However, the test data shows that the ratio initially starts at a high value, then drops rapidly as the loading is increased, until at 9,000 lb., the ratio bottoms out and then begins increasing with load all the way to failure.

SPECIMEN	HOLE STRAIN ε_o = 4,000 μin./in. μin./in.		GROSS STRAIN AT FAILURE μin./in. △5	
	PREDICTED	ACTUAL	PREDICTED	ACTUAL
75T060105-1001	16,632	19,958 △4	3,900	3,846
-1003	15,237	△2	4,299	4,039
-1005-1	11,430	9,904	4,051	5,316
-1005-2	11,430	△2	4,051	4,893
-1005-3	11,430	11,246	4,051	5,358
-1007	13,320	12,682	3,476	4,291
-1011	11,160	10,025	3,992	4,540
-1013	9,540	8,825	4,670	4,758
-1015-1	8,640	8,806	5,156	4,774
-1015-2	8,640	8,103	5,156	4,890
-1017	△1	7,047	△1	6,233
-1019-1	10,440	10,627	6,724	5,712
-1019-2	10,440	9,338	6,724	5,604
75T060106-1001	16,750	△3	3,910	△3
-1003	16,212	△3	4,041	△3
-1005	11,593	12,200 4	4,019	3,991
-1007-1	13,320	14,207 4	3,476	3,829
-1007-2	13,320	14,305 4	3,476	3,844
-1009	10,265	9,905 4	4,670	3,872
-1011-1	10,374	11,976	4,295	4,050
-1011-2	10,374	11,560	4,295	3,995
-1011-3	10,374	12,322	4,295	4,325
-1013	△1	14,436	△1	4,178
-1015	△1	11,505	△1	4,042
-1017	11,340	14,075 △4	4,083	3,828
-1019	13,608	11,362	4,815	5,806
-1021-1	15,372	11,256	4,785	6,306
-1021-2	15,372	10,621	4,785	5,814
-1021-3	15,372	10,417	4,785	5,832

△1 NOT ANALYZED BY COMPLIANCE MODEL
△2 STRAIN GAGE FAILED
△3 SPECIMEN NOT FABRICATED
△4 EXTRAPOLATED

△5 AVERAGE OF GAGES 1, 2, 5 & 6; SEE FIGURE 3.3, OR AVERAGE OF GAGES 1 & 2; SEE FIGURE 3.4.

TABLE 4.3 COMPARISON BETWEEN COMPLIANCE MODEL PREDICTIONS AND TEST RESULTS

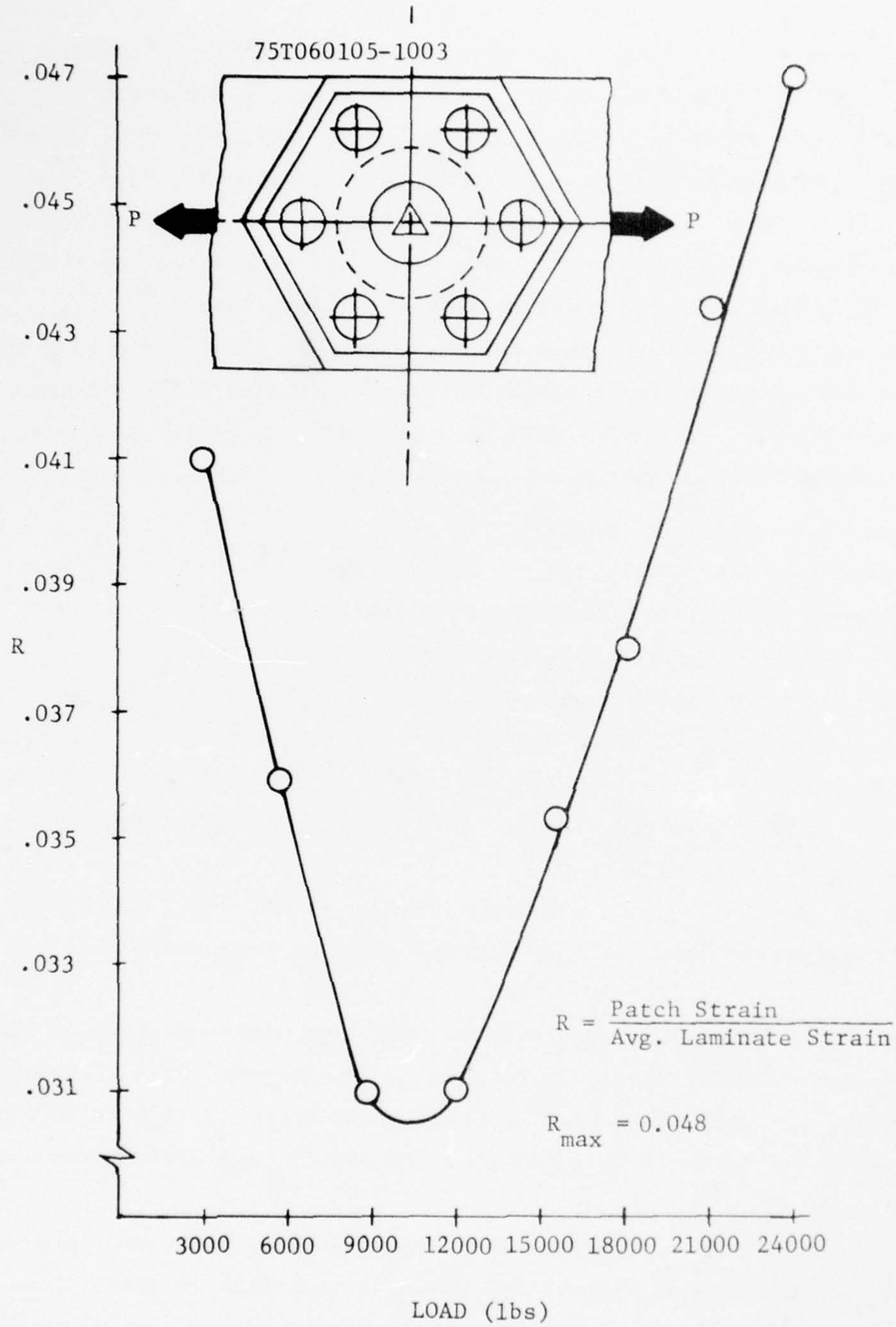


FIGURE 4-1 PLOT OF RATIO OF REPAIR PATCH STRAIN/GROSS LAMINATE STRAIN VS. LOAD FOR 75T060105-1003

Initially, due to fastener clearances, the only load transferred to the patch is through friction. Consequently the patch strain increases at a lower rate than the laminate strain. This causes a decrease in the strain ratio (R) with increasing load. At 9,000 lb the fasteners begin to load the patch. As fastener clearances are gradually overcome by strain, the strain ratio begins to increase with load. When all clearances are removed the strain ratio should remain constant with increasing load.

Figure 4-1 shows positive slope up to the failure point indicating that the specimen failed before the fastener clearances allowed fully effective loading on the patch. This inferrence is supported by a review of the test data. The specimen failed in the net section and very little strain (200 μin./in.) was measured on the patch.

Such behavior is characteristic of repair concepts with a relatively small axial separation between fastener rows due to the preload required to overcome the initial clearance. On the other hand, repair concepts with a larger axial separation between fastener rows are less affected by clearance. In Figure 4-2, the change in the slope of the curve shows that the 75T060105-1013 specimen clearance effects have been partially overcome at the moment of failure. The lack of an initial decrease in strain ratio above 7,000 pounds is not evident in this test.

In contrast to the reasonable agreement noted on all other specimens, the hole strain predictions for the 75T060106-1021 were overestimated by nearly 65%.

The 4-bolt 75T060106-1021 specimen was even more anomolous because its performance was virtually identical to that of the larger 8-bolt 75T060106-1019 specimen. The additional four fasteners were expected to provide a 15% increase in strain capability; however, no significant differences were noted.

Test records reveal no adequate explanation; in fact, the results are contradictory. As shown by Figures 4-3 and 4-4, the patch to laminate strain ratio at failure for the 8-bolt repair was almost 80% higher than the strain ratio for the 4-bolt specimen. This implies that the former specimen could be loaded to a higher strain level before failing at the hole. Yet, it failed through the net section at a lower strain level than the 4-bolt repair. The

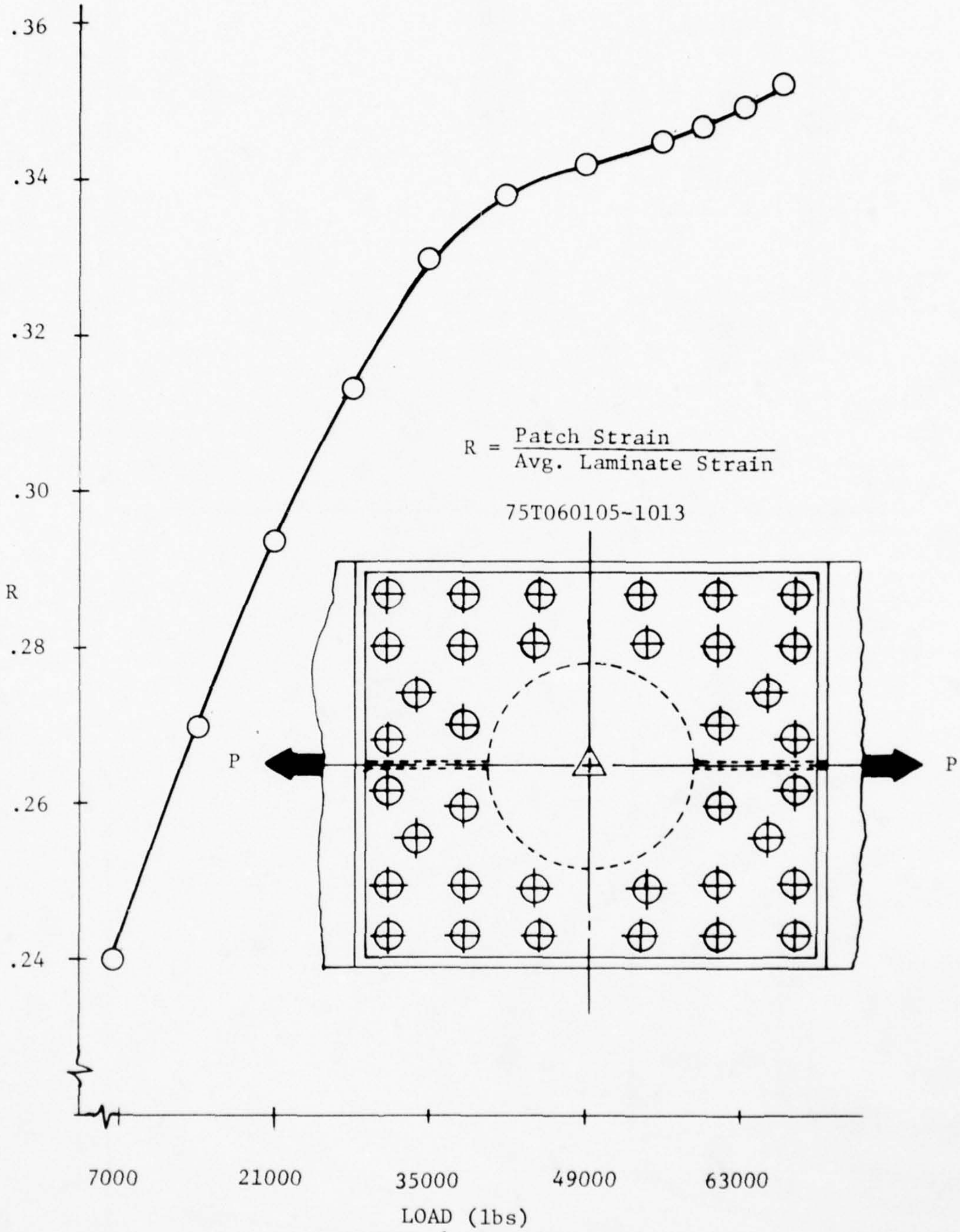


FIGURE 4-2 PLOT OF RATIO OF REPAIR PATCH STRAIN/GROSS LAMINATE STRAIN VS. LOAD FOR 75T060105-1013

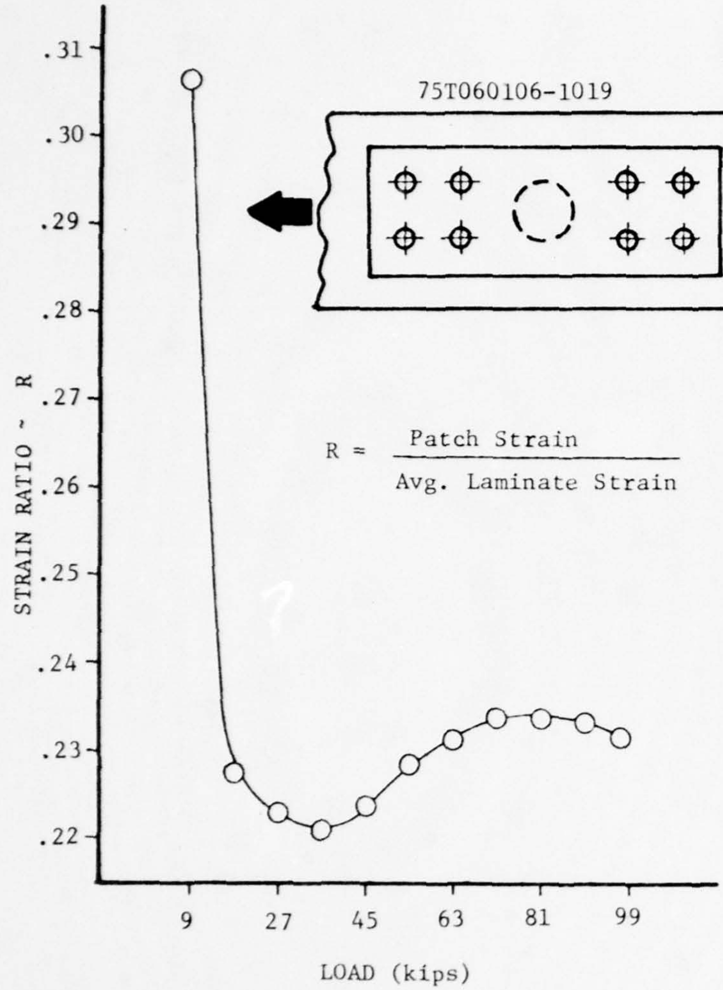


FIGURE 4-3 STRAIN RATIO VS LOAD FOR 75T060106-1019 SPECIMEN

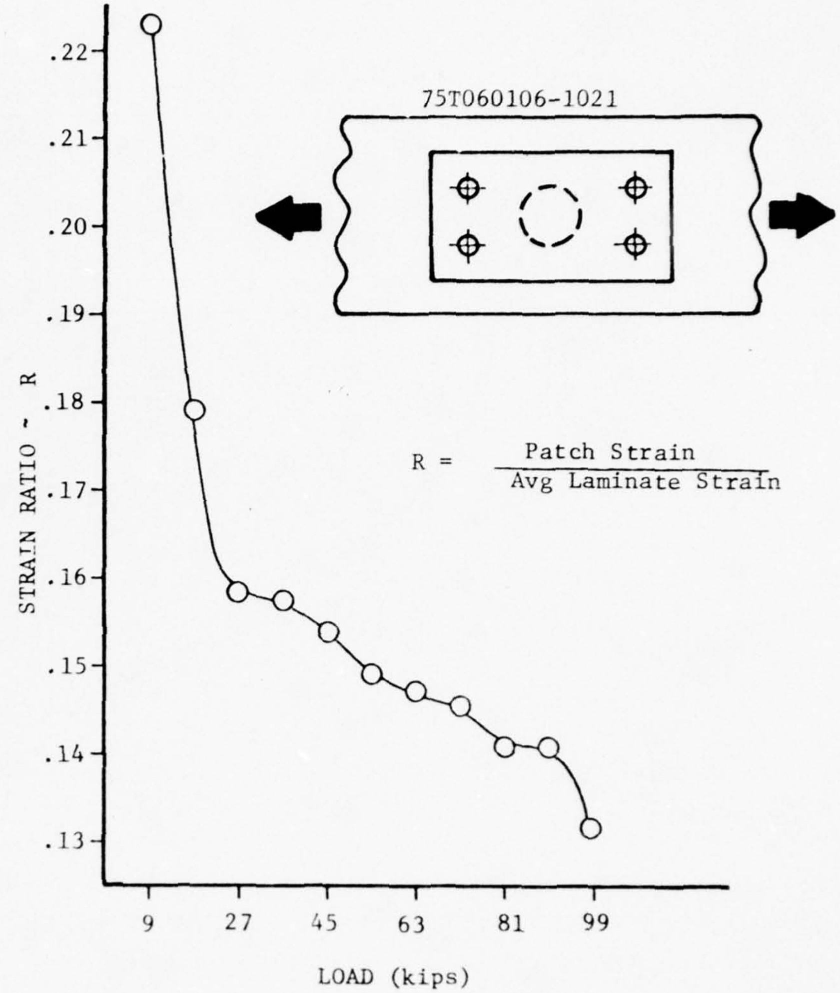


FIGURE 4-4 STRAIN RATIO VS LOAD FOR 75T060106-1021 SPECIMEN

4-bolt repair failed at a fastener line. The 8-bolt specimen geometry possibly prevented the outermost fasteners from being effectively loaded due to their proximity to the load pad buildups. Additional testing of these two repair designs is necessary before adequate conclusions can be made.

Table 4.3 shows generally good agreement exists between the predicted and actual gross failure strains. The analysis predicts the failures will initiate at the edge of the damage hole for all specimens. The 1/2 in. laminates all failed in this manner. The 3/16 in. laminates, except for the initial 1.0 hole designs, all failed at the outermost row of fasteners. This can be attributed to the magnification of bearing stress caused by rigid body tilting of the fasteners. This tilting occurs in the 3/16 in. panels because the 0.040 in. backing plates are ineffective in load transfer and cause the fasteners to be loaded in single shear.

Table 4.4 summarizes the hole strain at failure for the specimens which failed at the hole. The average failure strains for the 1.00, 2.50 and 4.00 in. holes agree quite closely with the maximum allowable hole strains predicted. (The predicted strains shown previously in Table 2.2 are included in Table 4.4 for convenience.) This agreement verifies that the role of size in reducing the strength of the laminate can be explained using the "characteristic dimension" theory (Ref. 6) and secondly, that a rational failure criterion exists for designing repairs.

SPECIMEN	HOLE DIA INCH	HOLE STRAIN AT FAILURE μin./in.	AVG HOLE STRAIN AT μin./in.	PREDICTED HOLE EDGE FAILURE STRAIN μin./in. (from Table 2-2)
75T060105-1003 75T060106-1021-1 -1021-2 -1021-3 -1019	1.0	19,914 17,745 14,879 14,925 16,493	16,791	16,380
75T060106-1005 -1007-1 -1007-2 -1017	2.5	12,060 13,168 13,748 13,522	13,125	11,575
75T060106-1009 -1011-3	4.0	9,579 13,323	11,451	11,138

TABLE 4.4 SUMMARY OF MEASURED HOLE STRAIN AT FAILURE VS. HOLE DIAMETER

5. CONCLUSIONS AND RECOMMENDATIONS

The program has demonstrated that mechanically attached patches that are designed for installation by field personnel can be used to repair damaged graphite/epoxy laminates. With the exception of two early specimens, the repairs all produced strength improvements of at least 39% over the unrepaired panels. Also most repaired panels withstood over 4000 μin./in. strain. The program also demonstrated that the repair concepts can be designed and analyzed using rapid, simple techniques.

The improvement ratio has been shown to be a convenient parameter for adjusting differences in the performance of differing repair designs to account for specimen size. However, because of the limited scope of this program, tests were not performed to determine (1) the actual baseline strength of the undamaged laminate (2) the strengths of the unrepaired laminates. While the predicted strengths employed in this study for the unrepaired laminates show good agreement with the measured failure strains at the hole edge, future test programs should include these additional specimens in order to precisely calibrate the effectiveness of each design.

Development efforts directed toward major task areas outlined below are recommended on the basis of the essential first step provided by this program in the area of field repairs of composite structures.

1. At the field repair level:
   - (a) Wing substructural involvement
   - (b) Assessment of strength and need to repair damaged structures
   - (c) Fatigue strength of repairs
   - (d) Feasibility of limited bonding
   - (e) Feasibility of composite repair patches
   - (f) Effect of laminate thicknesses, other than 3/16 and 1/2 inch thick, and different layups on design of repairs
2. At the depot repair level:
   - (a) Repair of sinewave graphite/epoxy substructure
   - (b) Structural beef-up
   - (c) Repair of large wing skin areas
   - (d) Improved bonding techniques
   - (e) NDT methods of damage assessment

6. COMPOSITE STRUCTURAL REPAIR WORK PACKAGE

The following paragraphs detail the procedures for installing a bolted repair patch on a damaged graphite/epoxy laminate.

It is imperative that the safety procedures and disposal methods be followed closely to minimize personal and environmental hazards.

When a damaged composite skin is to be repaired it is essential that the damage area size be established by a reliable form of Non-Destructive Testing (NDT). The total damage to the composite is often not accurately assessable visually.

The repairs discussed herein are only for wing torque box skins and cannot be used when substructural elements are involved.

6.1. SAFETY PROCEDURES

This defines the safety procedures that must be followed when working with graphite/epoxy laminates. The major wing structure elements are made of these materials. Personnel should be thoroughly familiar with these procedures before starting any repairs outlined in the following subordinate paragraphs.

WARNING

Health hazards related to graphite/epoxy laminates have not been determined. Gloves shall be worn when handling damaged structure. If an exposed graphite filament should penetrate the gloves and enter the skin, remove it with tweezers. A water filtered vacuum cleaner shall be used to remove and contain dust particles. A dust respirator and eye protection shall be worn when drilling, routing, grinding, or sanding graphite/epoxy laminates.

Graphite fibers have high electrical conductivity and can cause malfunction if allowed to get inside unsealed or unprotected electrical equipment.

6.2 DISPOSAL

Clean up and disposal of fibrous graphite and graphite/epoxy scrap shall be accomplished per the following four classes of waste material:

I. Non-charred laminate composites, and scrapped parts.

II. Pieces of laminate composite, accidently burned in a fire, or any other laminate material in which excessive heat has degraded the ability of the resin to hold graphite fibers in place.

III. Single fibers, fiber clumps, fragments, or other small fibrous scrap which could be easily transported by air currents or mechanical agitation.

IV. Nonfibrous dust (particles length 0.025 or less) resulting from drilling and routing operations.

CAUTION

Under no circumstance, shall any class I, II or III material be incinerated or placed in a waste container scheduled for incineration.

Use the following procedure for clean up and disposal of waste material.

I. Class I material will be placed in a waste container scheduled for dumping in a landfill.

II. Class II material shall be placed in a polyethylene bag or wrapped in polyethylene film (if too large to bag), the polyethylene shall be sealed to prevent release of graphite fibers into the environment, and the material shall be placed in a waste container scheduled for dumping in a landfill.

III. Class III material shall be cleaned up by first vacuuming the area and then sweeping up remaining fragments. The waste thus collected shall be disposed of same as class II material.

IV. Class IV material may be cleaned up and disposed of following standard procedures for general production waste material.

6.3 CLASSIFICATION OF DAMAGE

Damage to graphite/epoxy laminated structure will be classified according to thickness, damage location, and the diameter of a circle which encloses the damage.

6.3.1 WING SKIN DAMAGE

a. Determine the extent of the broken and delaminated skin using non-destructive inspection methods per NAVAIR 01-AV8A-3-7 SWP 007 08B. Mark outline of damage on the surface being repaired.

b. Determine the smallest circle which will totally enclose the damaged area.

c. Measure thickness of the skin in damaged area by ultrasonic inspection or mechanically by micrometer.

NOTE: The repairs described herein apply to 3/16 inch and 1/2 inch laminates.

d. Select patch and backing plate configuration from Figure 6-1 to match hole size indicated by NDT (steps a & b). When damage hole size falls between patch sizes the larger patch must be used.

NOTE: Repair patches and backing plates are defined in Figures 6-2 and 6-3.

	1.0 in. Hole	2.5 in. Hole	4.0 in. Hole
3/16 in. Laminates	75T060104 -2011 Patch -1019 Backing Plate	75T060103 -2001 Patch -1001 Backing Plate	75T060103 -2013 Patch -1015 Backing Plate
1/2 in. Laminates	75T060104 -2013 Patch -1017 Backing Plate	75T060104 -2001 Patch -1005 Backing Plate	75T060104 -2007 Patch -1011 Backing Plate

FIGURE 6-1

REPAIR PATCH CONFIGURATION

6.4 REPAIR OF 1 TO 4-INCH HOLE IN 3/16-INCH OR 1/2-INCH LAMINATE

See Figures 6-2 and 6-3.

Damage to 3/16-inch thick graphite/epoxy laminated skin, which can be enclosed in a 1 to 4-inch circle, may be repaired using the mechanically fastened titanium alloy patches and backing plates shown in Figure 6-2.

Damage to 1/2-inch thick graphite/epoxy laminated skin, which can be enclosed in a 1 to 4-inch circle, may be repaired using the mechanically fastened titanium alloy patches and backing plates shown in Figure 6-3.

6.4.1 TOOLS AND EQUIPMENT

Safety Glasses or Eye Shield

Nylon Lab Coat

Nylon or Rubber Gloves

White Marking Pencils

Straight Edge

Drafting Compass

Scale or Ruler

Bushing, Slip Fit, 1-1/4 inches long, 0.2503 OD, 0.128 ID

Drill Bit, Similar to MCAIR Tool TFIM 25.0253-351

Drill Bit, Number 30

Drill Guide with 0.2504 ID Slip Fit Bushing

Drill Motor, 22,000 RPM, 3/8-inch Chuck

Fillet Fairing Tool, Similar to MCAIR Tool TD 596A

Respirator, Dust, Fine Filter

Router Bit, Similar to MCAIR Tool TD 595T-1, -2, -3, -4, -5, -6

Sheetmetal Holder, Similar to MCAIR Tool WNX-1/4

Vacuum Cleaner, Water Filter Type.

6.4.2 MATERIALS

Backing Plates (2), 75T060104-1019, 75T060103-1015 or 75T060103-1001
per Figure 6-2 for 3/16-in. laminates

Patches, 75T060104-2011, 75T060103-2001 or 75T060103-2013
per Figure 6-2 for 3/16-in. laminates

Backing Plates (2), 75T060104-1005, 75T060104-1011 or 75T060104-1017
per Figure 6-3 for 1/2-in. laminates

Patches, 75T060104-2001, 75T060104-2007 or 75T060104-2013
per Figure 6-3 for 1/2-in. laminates

Bolts, ST3M430V4-6AS, as required for 3/16-in. laminates

Bolts, ST3M430V4-13AS, as required for 1/2-in. laminates

Paper, abrasive, P-P-101, 80 grit and 240 grit

Sealing Compound, PR 1422, class A-2 (05027) Products
Research and Chemical Corp.

Wiping Materials:

Cheesecloth, bleached-CCC-C-440

Keybak Aerospace Wipers, 877, (86159) Chicopee Mills, Inc.

Rymplecloth, 301, (97327) The Kendall Company

1, 1, 1-Trichloroethane, 0-T-620

6.4.3 PROCEDURES

WARNING

Personnel working with graphite/epoxy laminates must be thoroughly familiar with the safety precautions in paragraphs 6.1 and 6.2.

a. Layout and cut the smallest diameter hole which can enclose the damage area. Use drill motor and 1/4-inch router bit. Use vacuum cleaner continuously to remove dust.

b. Check to see that backing plate appropriate for the repair can be inserted through the hole. If necessary slightly enlarge diameter of hole to allow insertation. Do not exceed hole diameter appropriate for patch being used.

c. Clean skin surface and remove paint from area under patch by sanding with 80 grit abrasive paper until almost all paint has been removed. Finish sand with 240 grit abrasive paper until all paint is removed. Be careful not to sand unnecessarily into laminate. Use vacuum cleaner to remove dust.

d. Determine the principal tension or compression load direction of laminate.

Using a pencil and straight edge, measure and layout two perpendicular centerlines of the damaged area. One centerline must coincide with principal load direction.

e. Center patch on damaged area. Align principal load direction with principal load direction centerline on patch. Pencil mark hole locations on the skin. Mark edges of patch so it can be returned to the same location.

f. Remove patch and check hole locations for 1/2-inch minimum edge distance.

g. Replace and align patch. Check fit of patch and backing plate against contour of skin. Form patch and backing plate as necessary to fit skin contour. Secure patch in place with masking tape.

h. Insert a .2503 OD bushing with a number 30 ID hole in any selected hole in the patch. See Figure 6-4.

i. Drill pilot hole in graphite laminate skin. Lubricate liberally with water.

j. Remove bushing and insert .2503 step pin which matches pilot hole (0.1285).

k. Slide drill guide with .2504 ID slip fit bushing over pin. Secure drill guide then remove step pin. See Figure 6-5.

l. Use TFIM 25.0253-351 drill with drill motor to open hole in the graphite laminate skin. Lubricate liberally with water. See Figure 6-6.

m. Replace drill guide and bushing with WNX-1/4 or equivalent sheetmetal holder.

n. Repeat steps h thru m until all holes are drilled.

o. Wipe area with cheesecloth, Rymplecloth, or Aerospace Wipers. Remove foreign matter from fuel cavity with vacuum cleaner. Clean patch with wiping material dampened with 1, 1, 1-trichloroethane.

p. Fay seal mating surfaces of backing plates and wing skin with PR 1422 sealing compound, using TD 596A fairing tool.

q. Place backing plates in position through hole in skin. Maintain alignment by installing 2 or 3 bolts in each plate finger tight. Clamp in place and allow sealant to cure sufficiently to hold backing plates in place.

r. Fay seal mating surfaces of patch and wing skin with PR 1422 sealing compound, using TD 596A fairing tool.

s. Locate patch on wing skin. Install ST3M430V4-6AS or ST3M430V4-13AS bolts wet with sealant. Start bolt threads in nut plates by hand.

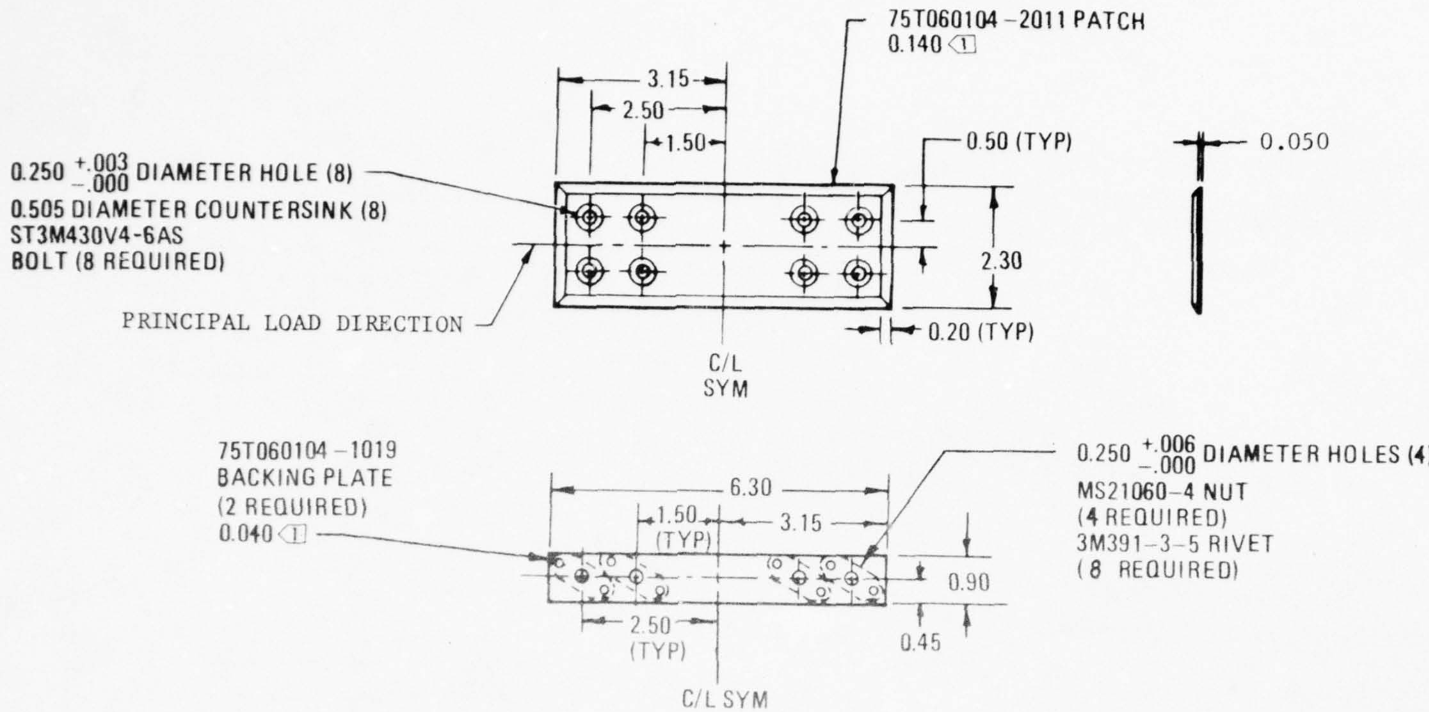
t. Tighten bolts using a diagonally opposite pattern.

u. Torque bolts 50-70 inch-pounds.

v. Remove excess sealing compound and fair patch with mold line skin.

w. Retorque bolts 50-70 inch-pounds.

x. Touch up paint per NAVAIR 01-AV8A-3-6.



NOTE

1 6AL-4V TITANIUM ALLOY, MIL-T-9046
TYPE III, COMPOSITION C, CONDITION ANNEALED

REPAIR OF 1.0 INCH HOLE IN 3/16 INCH LAMINATE

FIGURE 6-2 REPAIR OF 3/16 INCH LAMINATE

AD-A067 923 MCDONNELL AIRCRAFT CO ST LOUIS MO F/G 11/2

BOLTED FIELD REPAIR OF COMPOSITE STRUCTURES. (U)

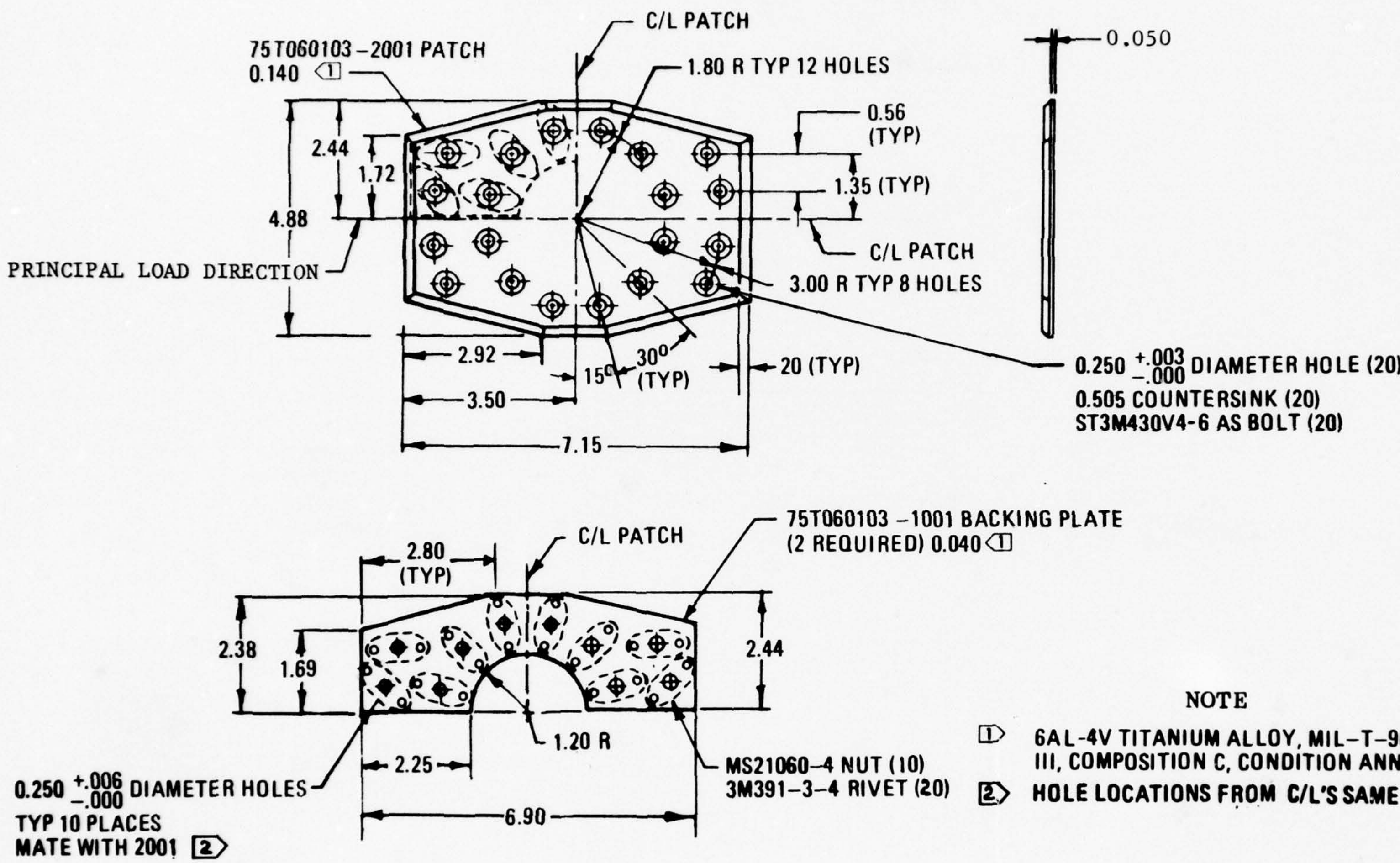
MAR 79 J B WATSON, D A GLAESER, F L HARVEY N62269-77-C-0366

UNCLASSIFIED MDC-A5583 NADC-77109-30 NL

2 OF 3

ADA

067923

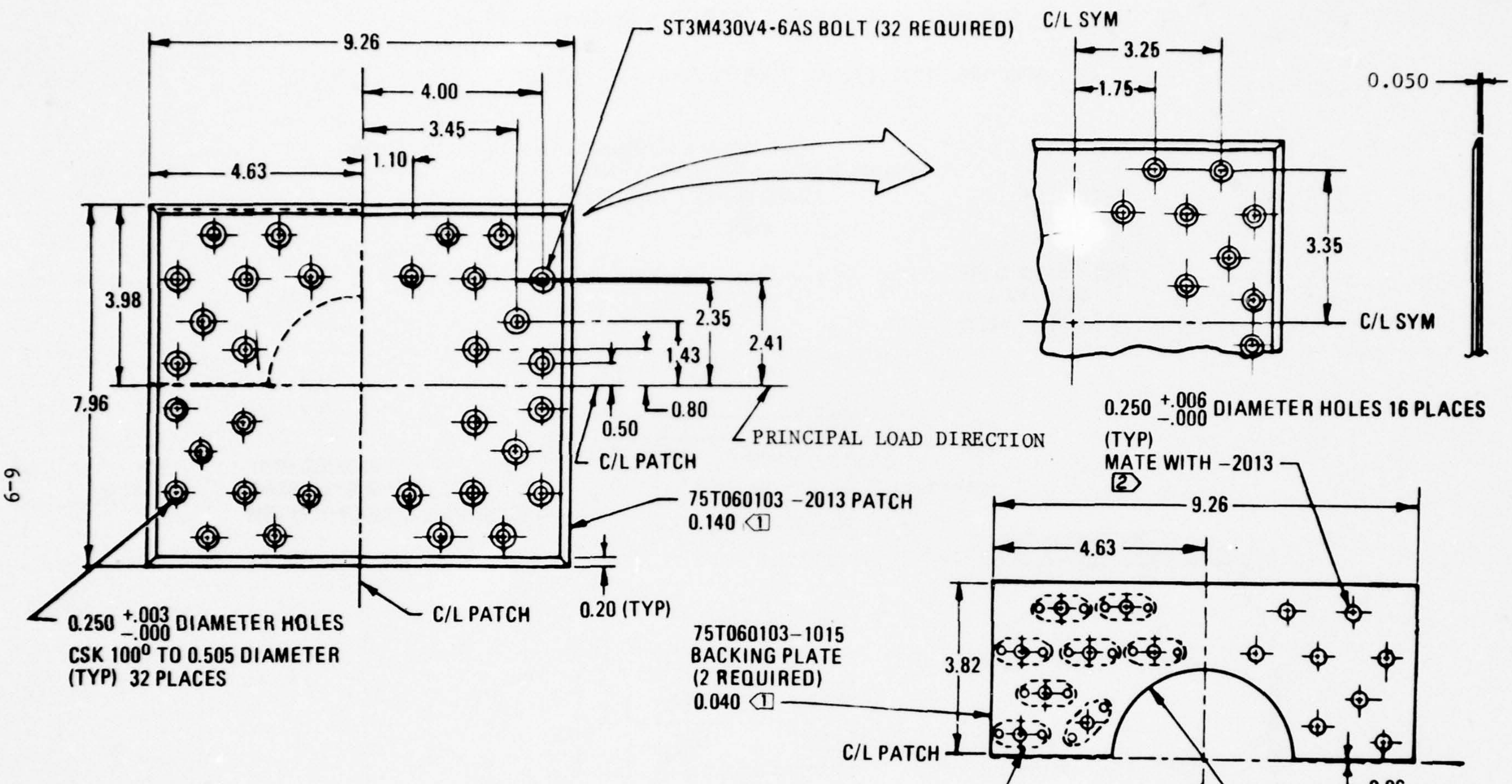


NOTE

1. 6AL-4V TITANIUM ALLOY, MIL-T-9046, TYPE III, COMPOSITION C, CONDITION ANNEALED.
2. HOLE LOCATIONS FROM C/L'S SAME AS -2001 PATCH

REPAIR OF 2.5 INCH HOLE IN 3/16 INCH LAMINATE

FIGURE 6-2 CONTINUED

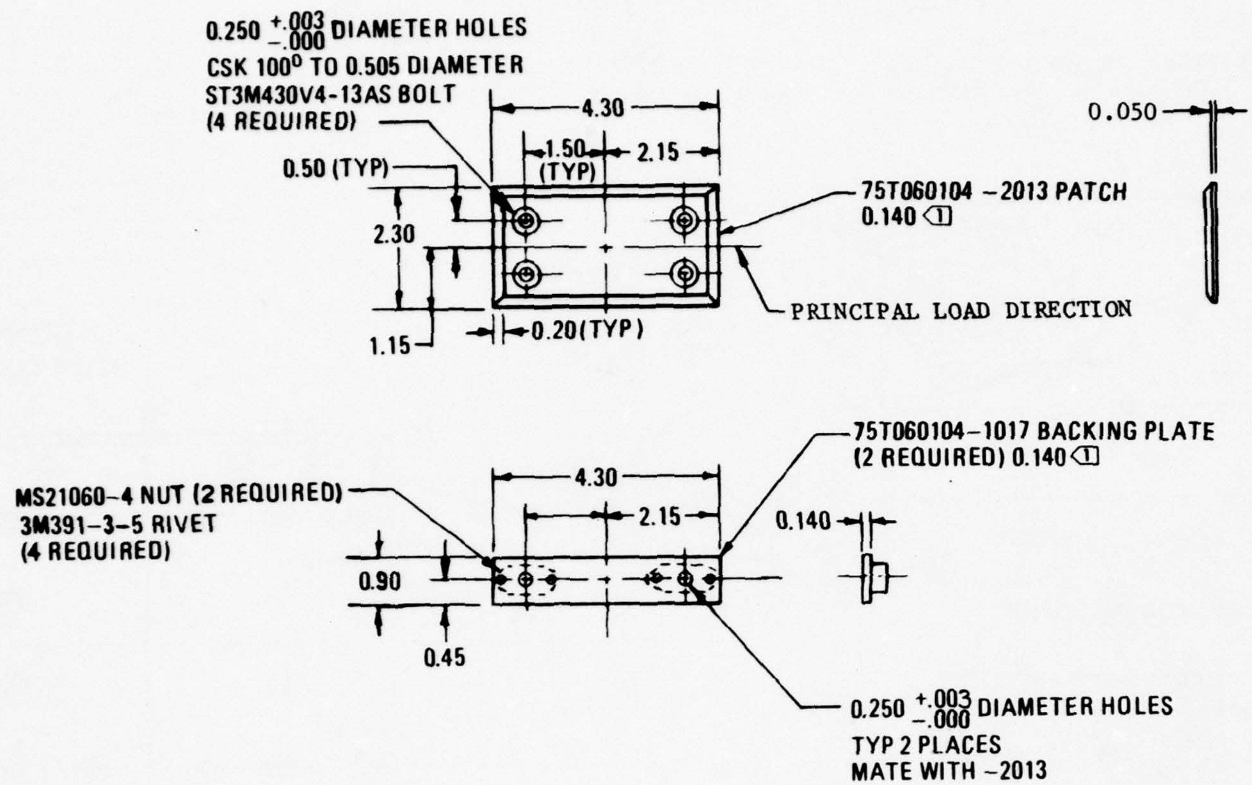


NOTE

1. 6AL-4V TITANIUM ALLOY, MIL-T-9046 TYPE III, COMPOSION C, CONDITION ANNEALED.
2. HOLE LOCATIONS FROM C/L'S SAME AS -2013 PATCH

REPAIR OF 4.0 INCH HOLE IN 3/16 INCH LAMINATE

FIGURE 6-2 CONTINUED

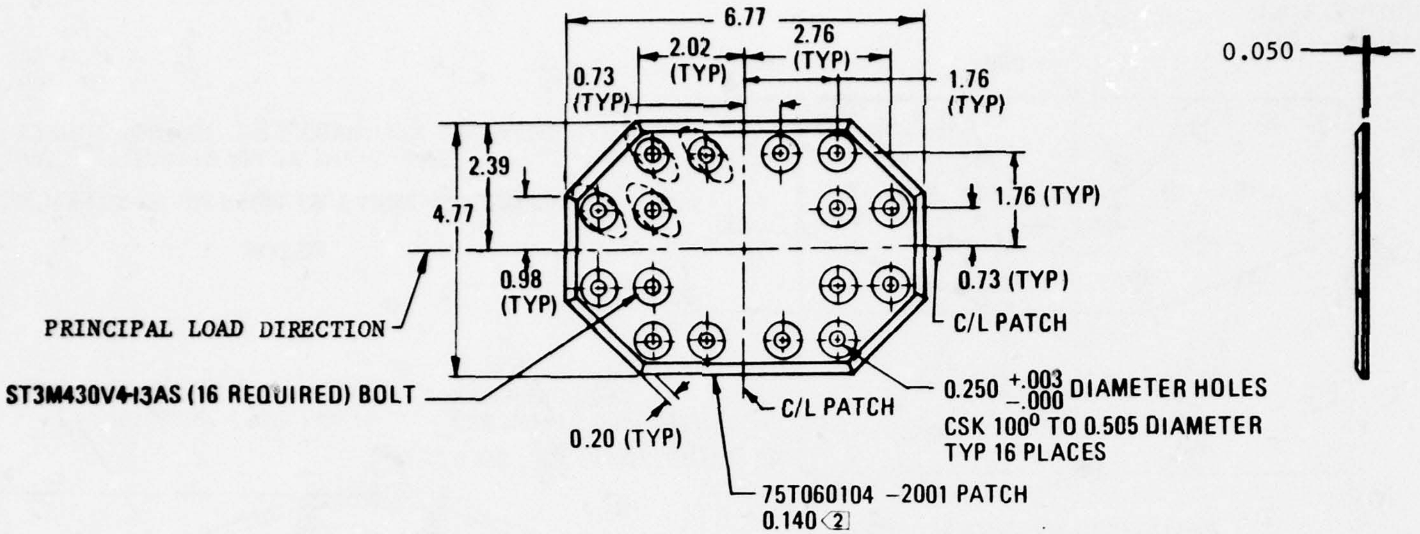


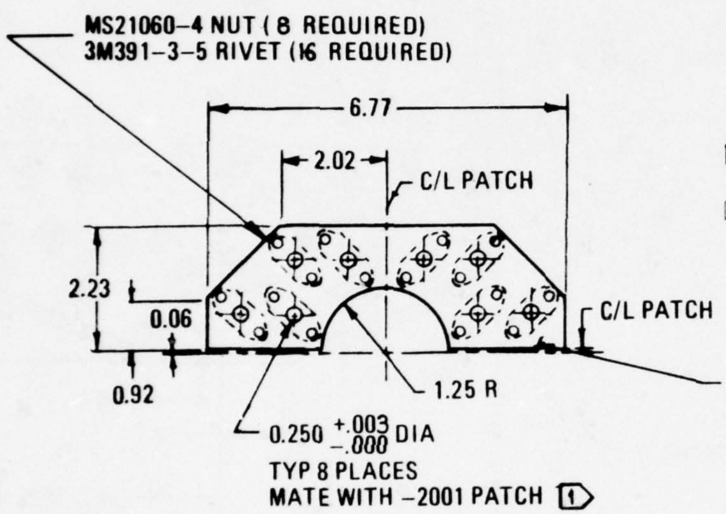
NOTE

1 6AL-4V TITANIUM ALLOY
MIL-T-9046, TYPE III, COMPOSITION C,
CONDITION ANNEALED

REPAIR OF 1.0 INCH HOLE IN 1/2 INCH LAMINATE

FIGURE 6-3 REPAIR OF 1/2 INCH LAMINATE



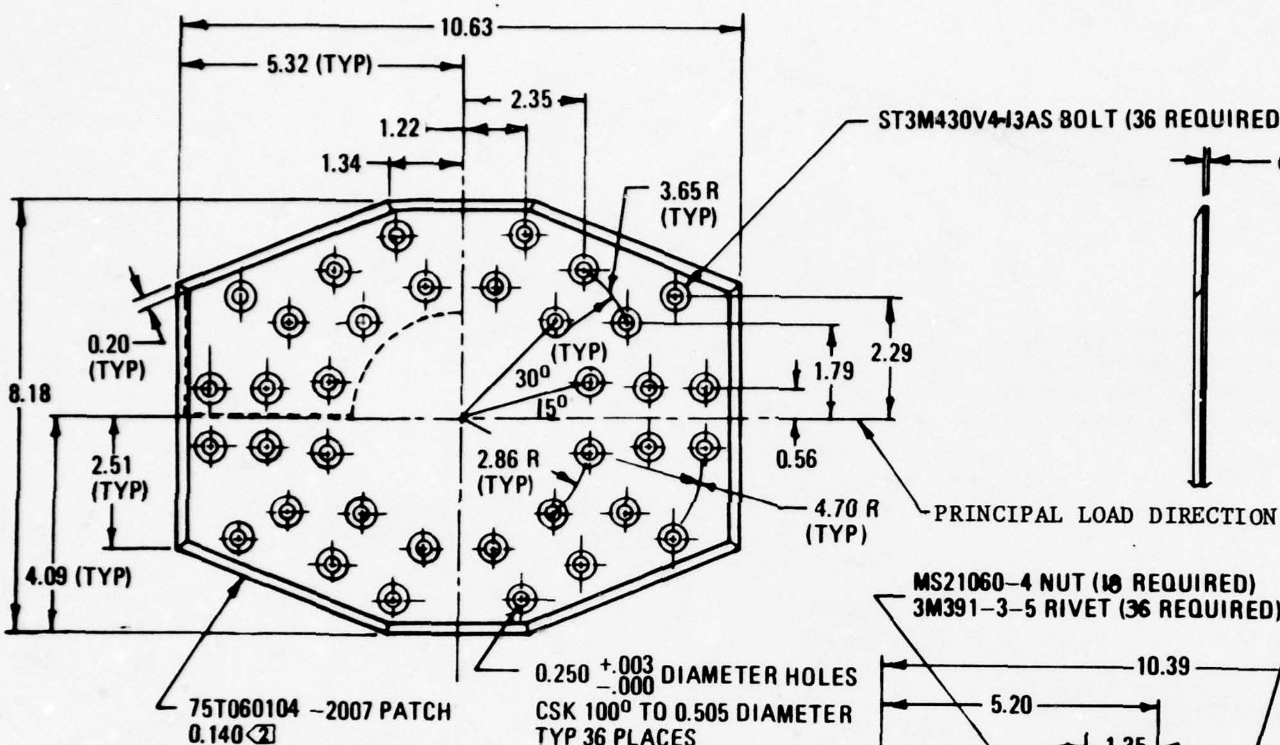


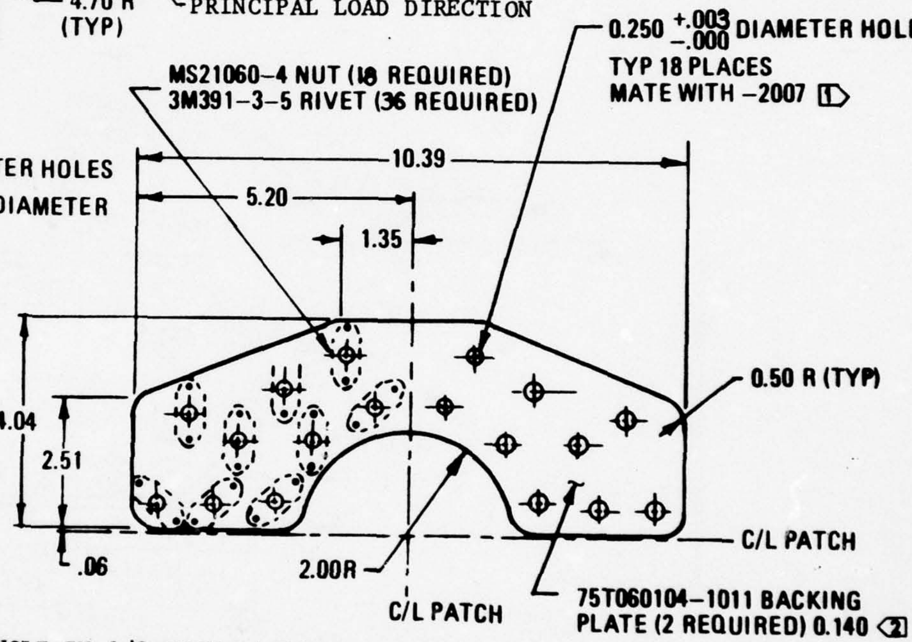
NOTES

1. HOLE DIMENSION FROM C/L'S OF BACKING PLATE SAME AS 2001 PATCH
2. 6AL-4V TITANIUM ALLOY, MIL-T-9046 TYPE III, COMPOSITION C, CONDITION ANNEALED

REPAIR OF 2.5 INCH HOLE IN 1/2 INCH LAMINATE

FIGURE 6-3 CONTINUED





NOTES

1. HOLE LOCATIONS FROM C/L'S SAME AS -2007 PATCH
2. 6AL-4V TITANIUM ALLOY, MIL-T-9046 TYPE III, COMPOSITION C, CONDITION ANNEALED

REPAIR OF 4.0 INCH HOLE IN 1/2 INCH LAMINATE

FIGURE 6-3 CONTINUED

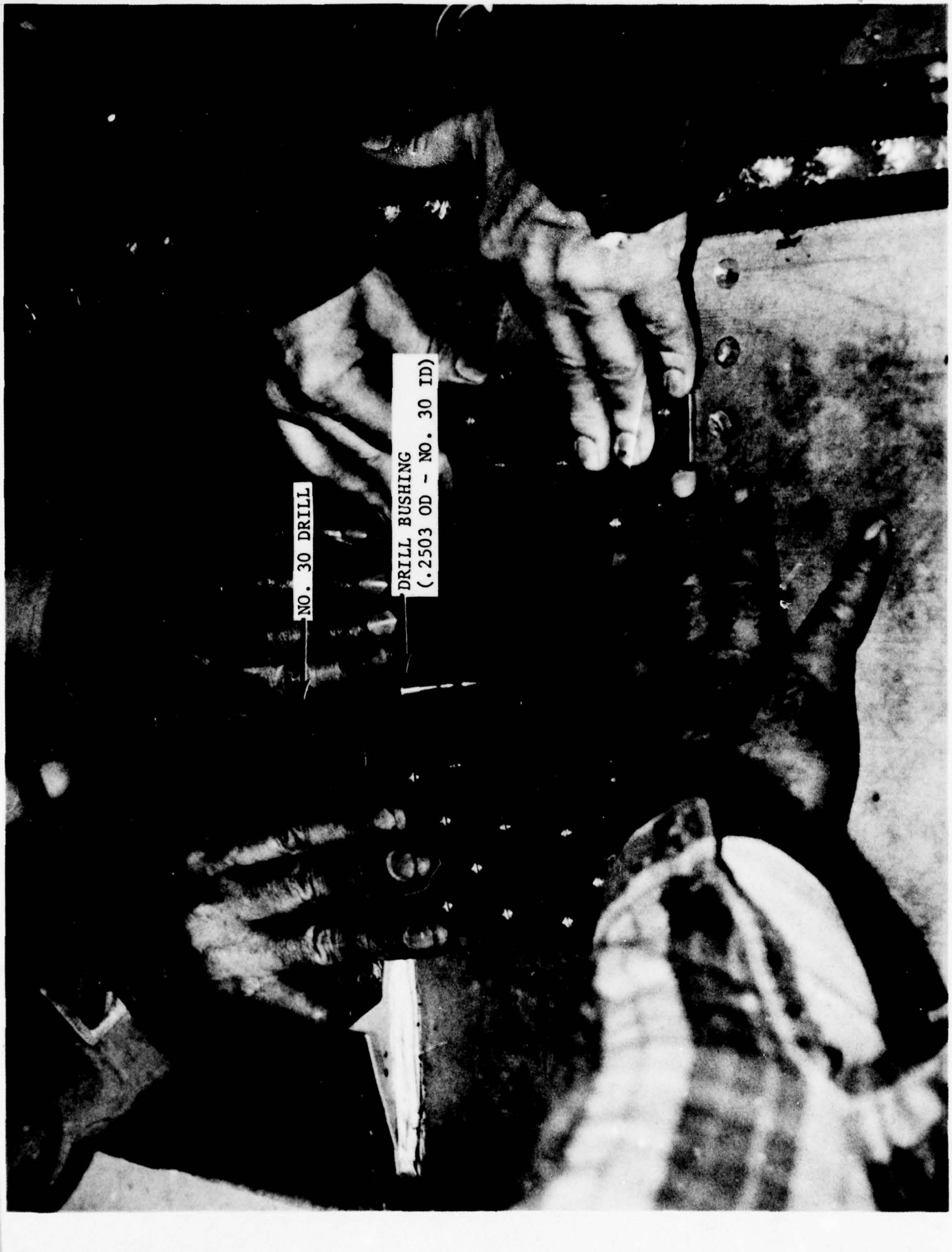


FIGURE 6-4 DRILLING PILOT HOLE

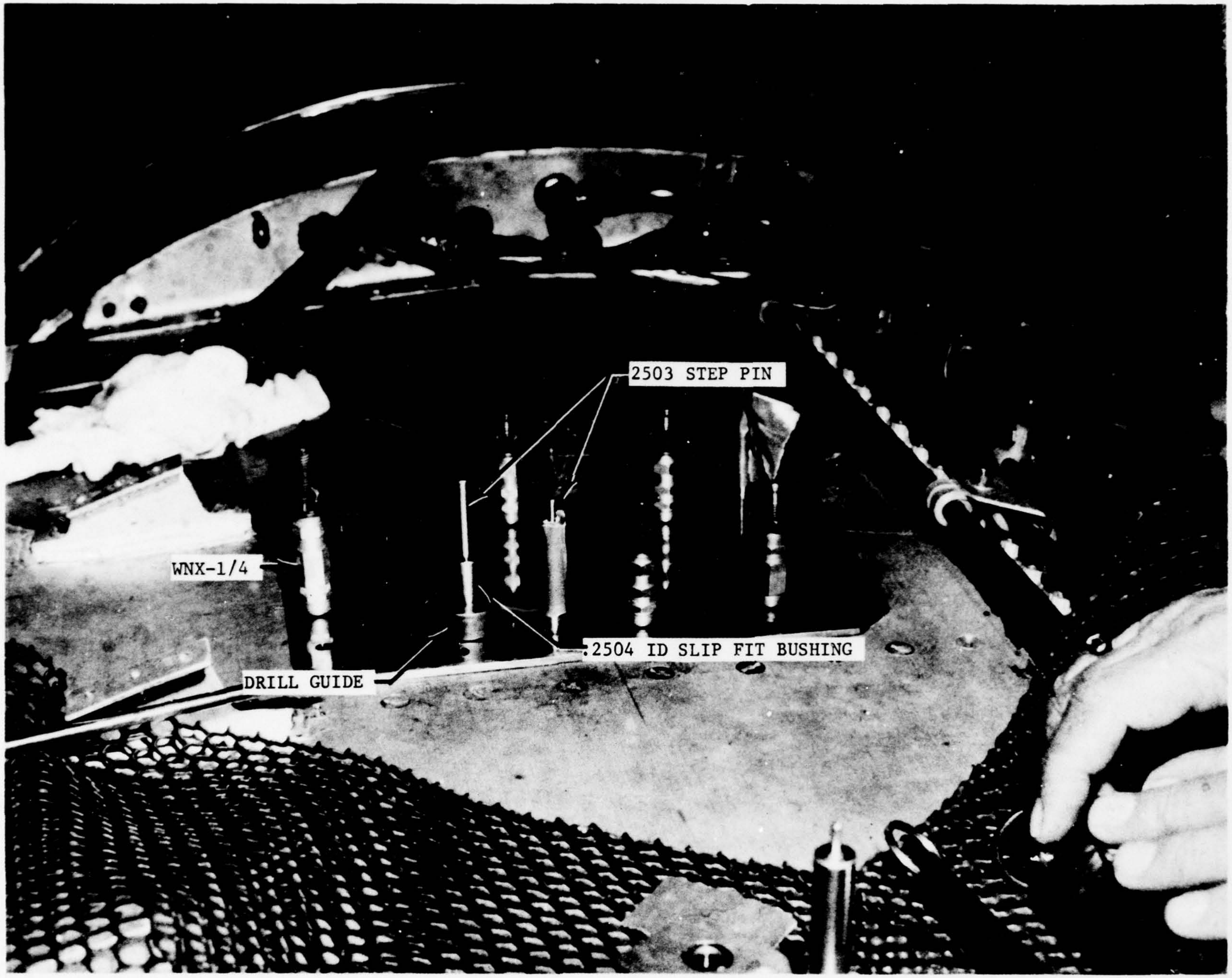


FIGURE 6-5 ALIGNMENT OF DRILL GUIDE

FIGURE 6-6 DRILLING ATTACHMENT HOLE IN LAMINATE

REFERENCES

1. K. V. Stenberg, et al., Advanced V/STOL Composite Wing Development Program, Final Technical Report, Contract N00019-76-C-0242, 30 November 1977.

2. F. Voight, The Load Distribution in Bolted or Riveted Joints in Light Alloy Structures, NACA TM 1135, April 1947.

3. S. Kekhnitskii, S., Theory of Elasticity of an Anisotropic Elastic Body, Holden-Day, 1965.

4. Advanced Composites Design Guide, Rockwell International, Volume II - Analysis, 1973.

5. W. Barrois, "Stresses and Displacements Due to Load Transfer by Fasteners", Engineering Fracture Mechanics, Volume 10, No. 1, 1978.

6. J. M. Whitney, and R. J. Nuismer, "Stress Fracture Criteria for Laminated Composites Containing Stress Concentrations", Journal of Composite Materials, Volume 8, 1974.

7. Okamura, et al, Applications of the Compliance Method to Fracture Mechanics, ASTM STP 536, 1973.

8. M. Greenspan, "Axial Rigidity of Perforated Structural Members", Journal of Research of the National Bureau of Standards, Vol. 31, December 1943.

9. Ernest E. Sechler, Elasticity in Engineering, Dover, 1952.

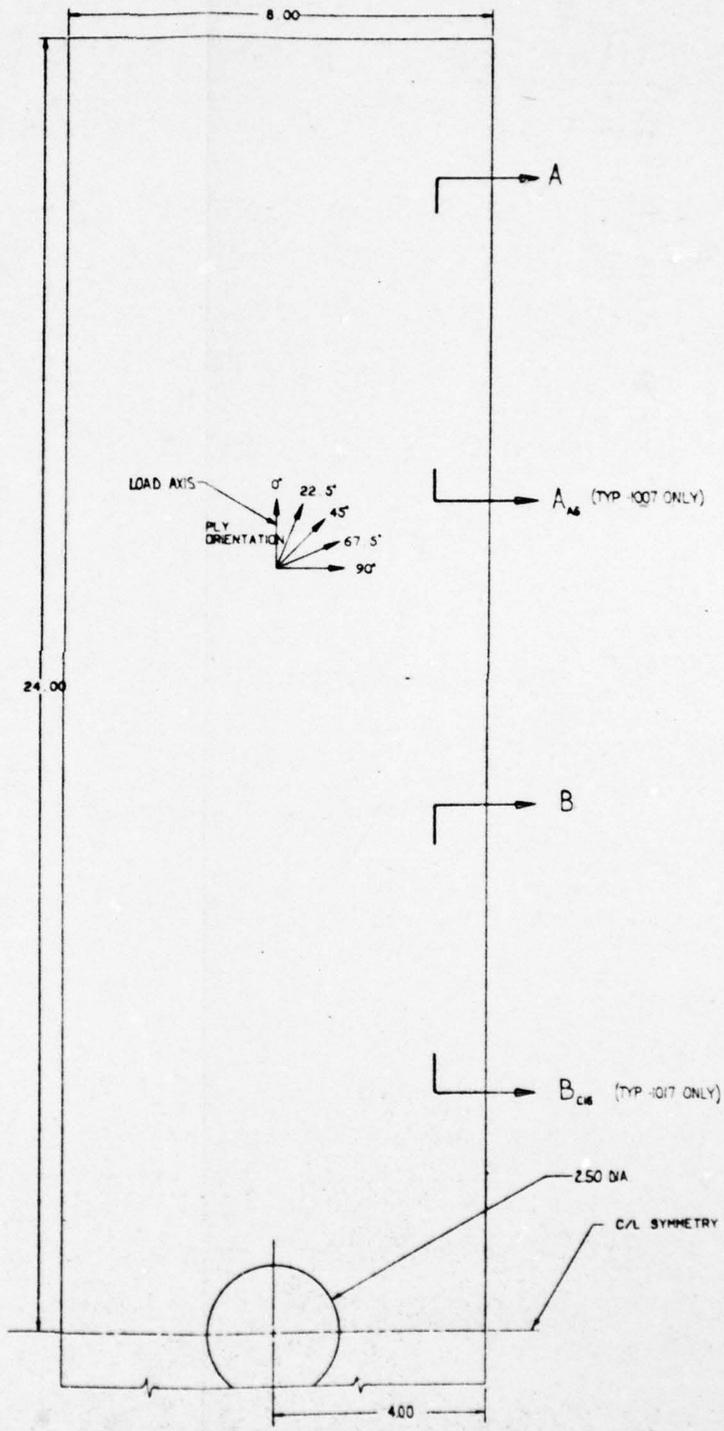
APPENDIX A

ENGINEERING DRAWINGS

APPENDIX A

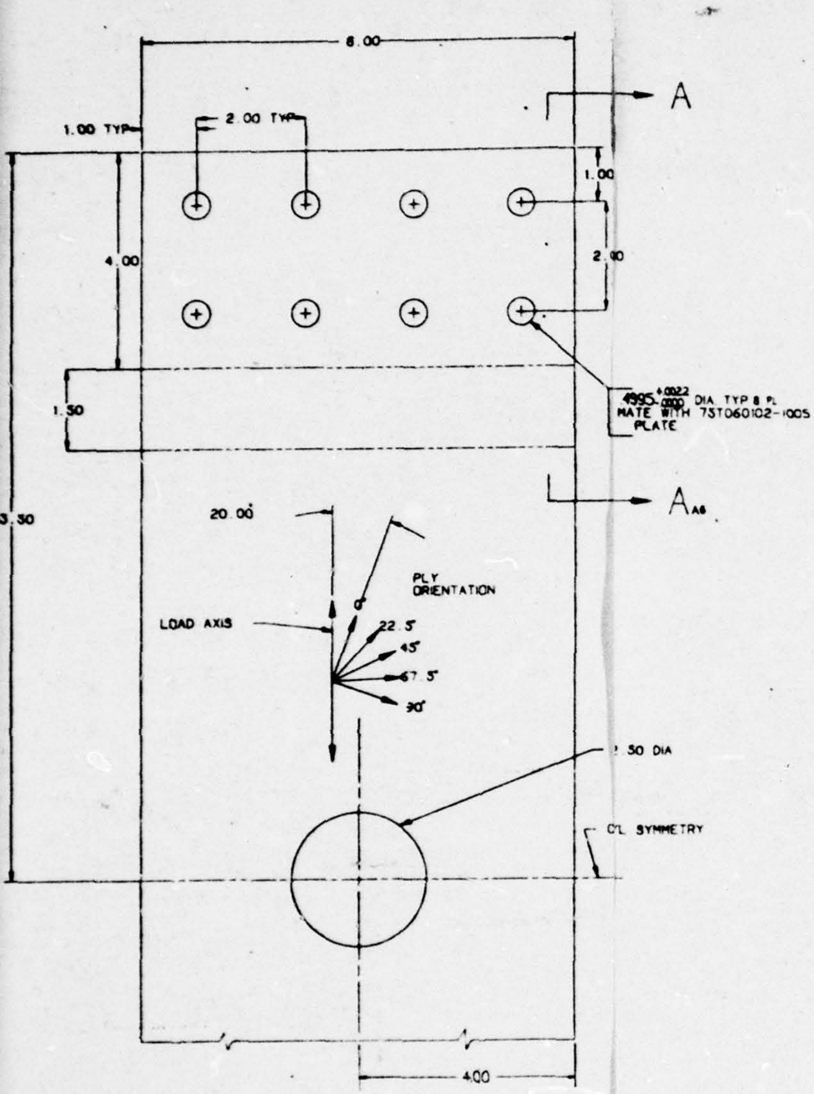
ENGINEERING DRAWINGS

The engineering drawings for the test specimens, load plates, repair patches and backing plates, repair assemblies and compression specimen edge supports are presented herein.

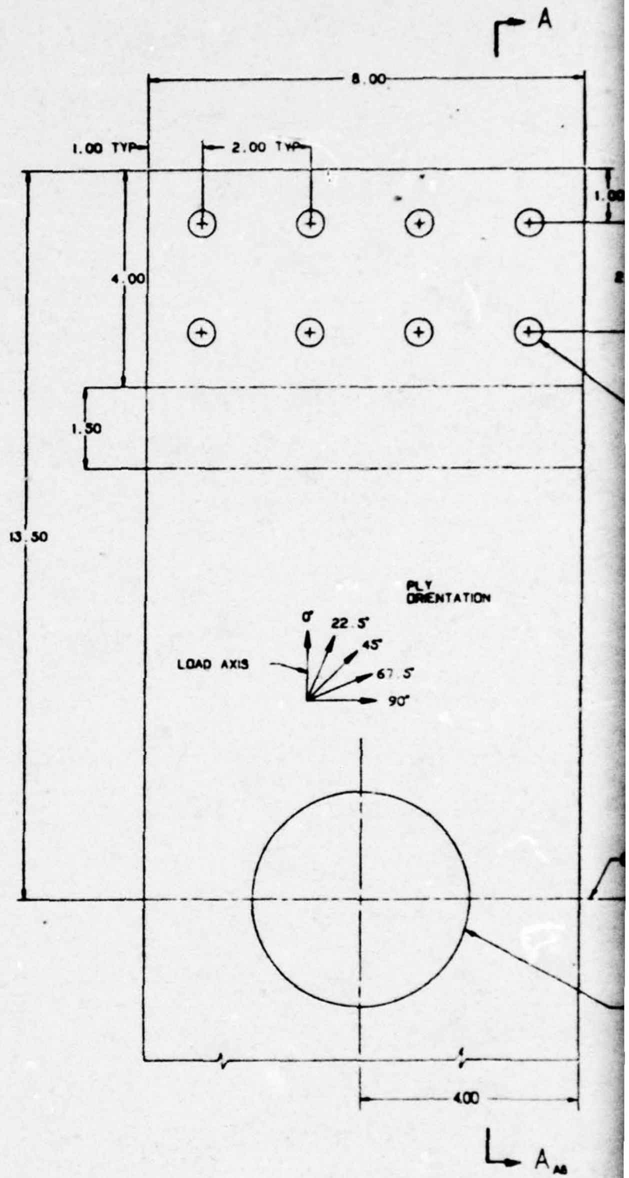


-1007 SPECIMEN (SEE SECTION A-A)
-1017 SPECIMEN (SEE SECTION B-B)

10 | 9 | 8



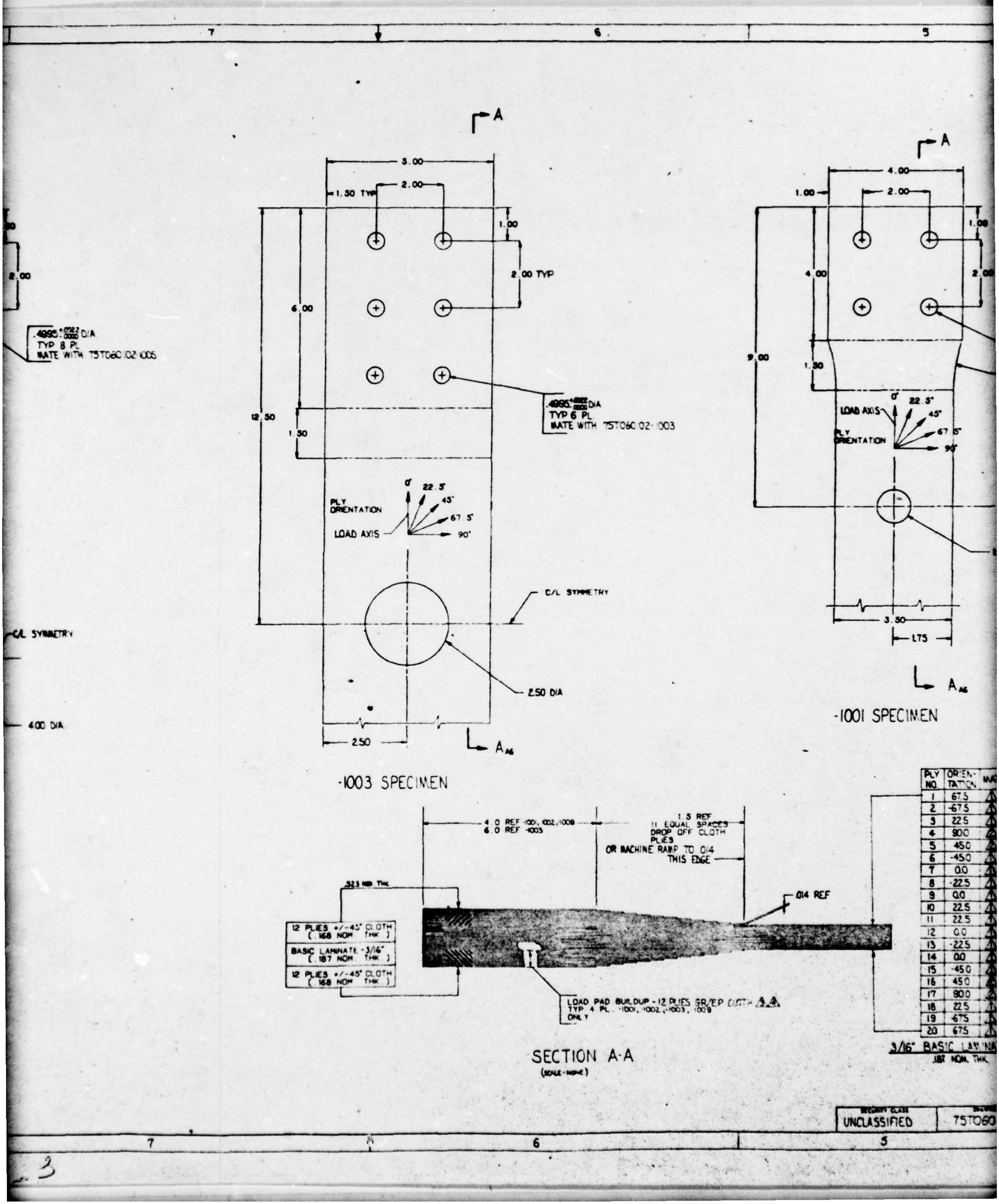
-1009 SPECIMEN



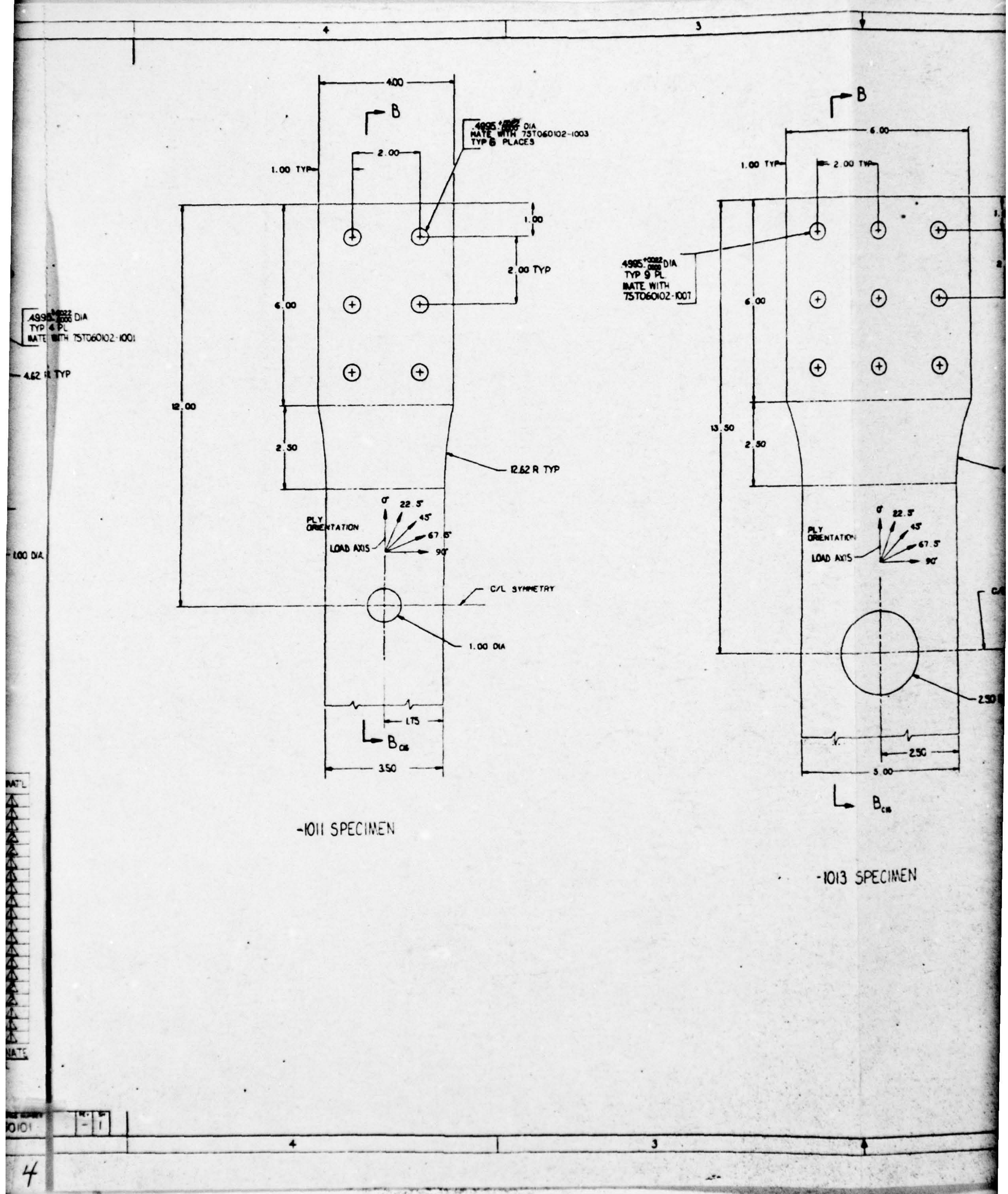
-1005 SPECIMEN

SECURITY CLASS	DRAWING NUMBER	REV	SH
UNCLASIFIED	75T060101		1

10 | 9 | 8

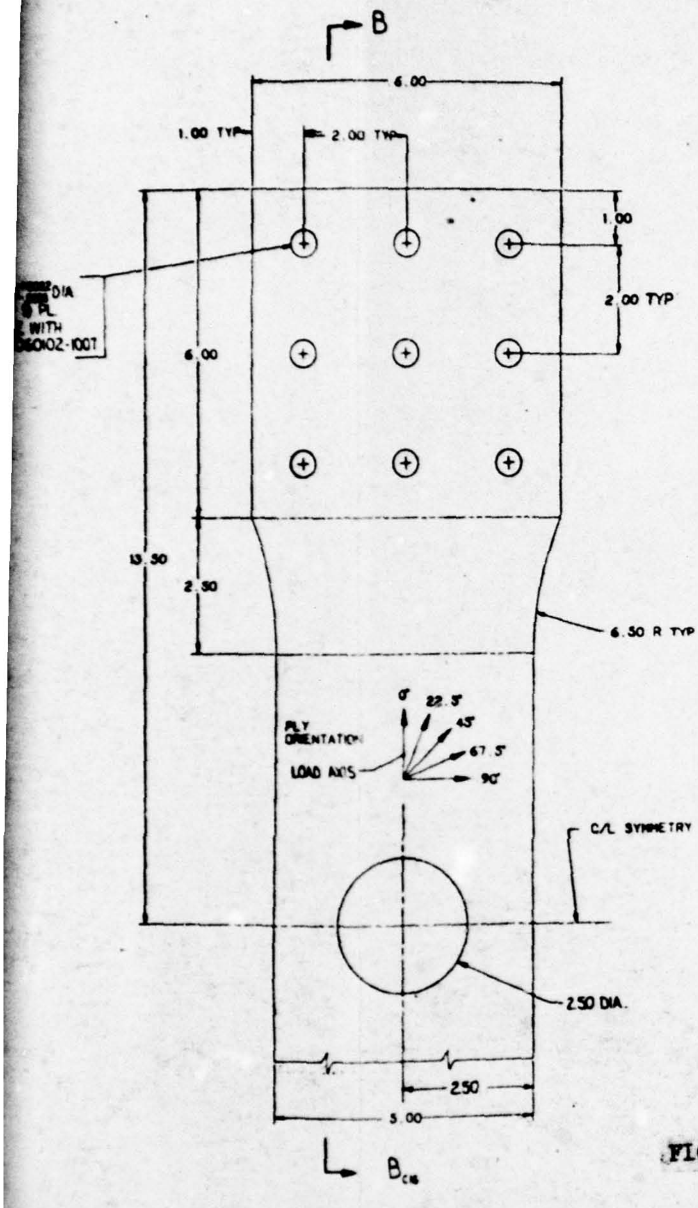


PLY NO.	ORIEN-TATION	MAT
1	67.5	
2	-67.5	
3	22.5	
4	90.0	
5	45.0	
6	-45.0	
7	0.0	
8	-22.5	
9	0.0	
10	22.5	
11	22.5	
12	0.0	
13	-22.5	
14	0.0	
15	-45.0	
16	45.0	
17	90.0	
18	22.5	
19	-67.5	
20	67.5	


4.00
B
.4995 DIA
MATE WITH 75T060102-1003
TYP 6 PLACES
2.00
1.00 TYP
1.00
2.00 TYP
6.00
12.00
2.50
12.62 R TYP
PLY ORIENTATION
LOAD AXIS
22.5°
45°
67.5°
90°
C/L SYMMETRY
1.00 DIA
1.75
3.50
-1011 SPECIMEN
TYP 4 PL
MATE WITH 75T060102-1001
4.62 R TYP
6.00
2.00 TYP
.4995 DIA
TYP 9 PL
MATE WITH 75T060102-1007
13.50
2.50
2.50
5.00
-1013 SPECIMEN
4
3

ZONE	LTR	DESCRIPTION	DATE	APPROVED
	A	SEE DCN A		

1. GRAPHITE EPOXY TAPE PER MMS 549 TYPE II (AS/3501-6) .0104 +/-.0006 CURED PLY THICKNESS.
2. GRAPHITE EPOXY TAPE PER MMS 549 TYPE I (AS/3501-6) .0052 +/-.0003 CURED PLY THICKNESS.
3. GRAPHITE EPOXY CLOTH PER MMS 544 (HMF 133/3501-6) .014 CURED PLY THICKNESS. THESE PLIES ORIENTATED AT +/-45 DEGREES.
4. CLOTH PLIES FOR LOAD PAD BUILDUPS TO BE PRECURED AND SECONDARILY BONDED TO PRECURED BASIC LAMINATE WITH EC2216 PER PS 11308. TYP ALL SPECIMENS EXCEPT -1007 AND -1017 (COMPRESSION SPECIMENS).
5. FABRICATE PER PS14240.
6. INSPECT PER PS 21233.
7. MARK PARTS PER PS160001.
8. MULTIPLE SPECIMENS MAY BE CUT FROM LARGER LAMINATE LAYUPS, PER PS 1418.
9. INDIVIDUAL PLIES ARE NOT GIVEN DASH NOS. SEE F/D ZONES A5, B7, B15 & C17 FOR LAYUP SEQUENCE AND ORIENTATION.



-1013 SPECIMEN

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FIGURE A-1 DETAILED DRAWING OF COMPOSITE REPAIR LAMINATES

A-2

QTY REQD	QTY REQD	QTY REQD	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE	DRAWING OR SPECIFICATION NO	NOTE NO	ZONE
AS REQ'D			EC 2216	ADHESIVE				4	
38	40	12	-1019	1/2 SPECIMEN					
–	40	12	-1017						
38	40	12	-1015						
38	40	12	-1013						
38	4	12	-1011	1/2 SPECIMEN					
24	16	4	-1009	3/16 SPECIMEN					
–	16	4	-1007						
24	16	4	-1005						
24	16	4	-1003						
24	16	4	-1001	3/16 SPECIMEN					
			9	CLOTH PLY		3			
			9	10 MIL TAPE PLY		1			
			9	5 MIL TAPE PLY		2			

PARTS LIST

DASH NO	FIN QTY REQD	NEXT ASSY	USED ON
-1013	4	75T060105	TEST
-1011	4	75T060105	TEST
-1009	1	75T060105	TEST
-1007	1		TEST
-1005	4		TEST
-1003	4		TEST
-1001	4	75T060105	TEST

CONT'D SH 2

APPLICATION

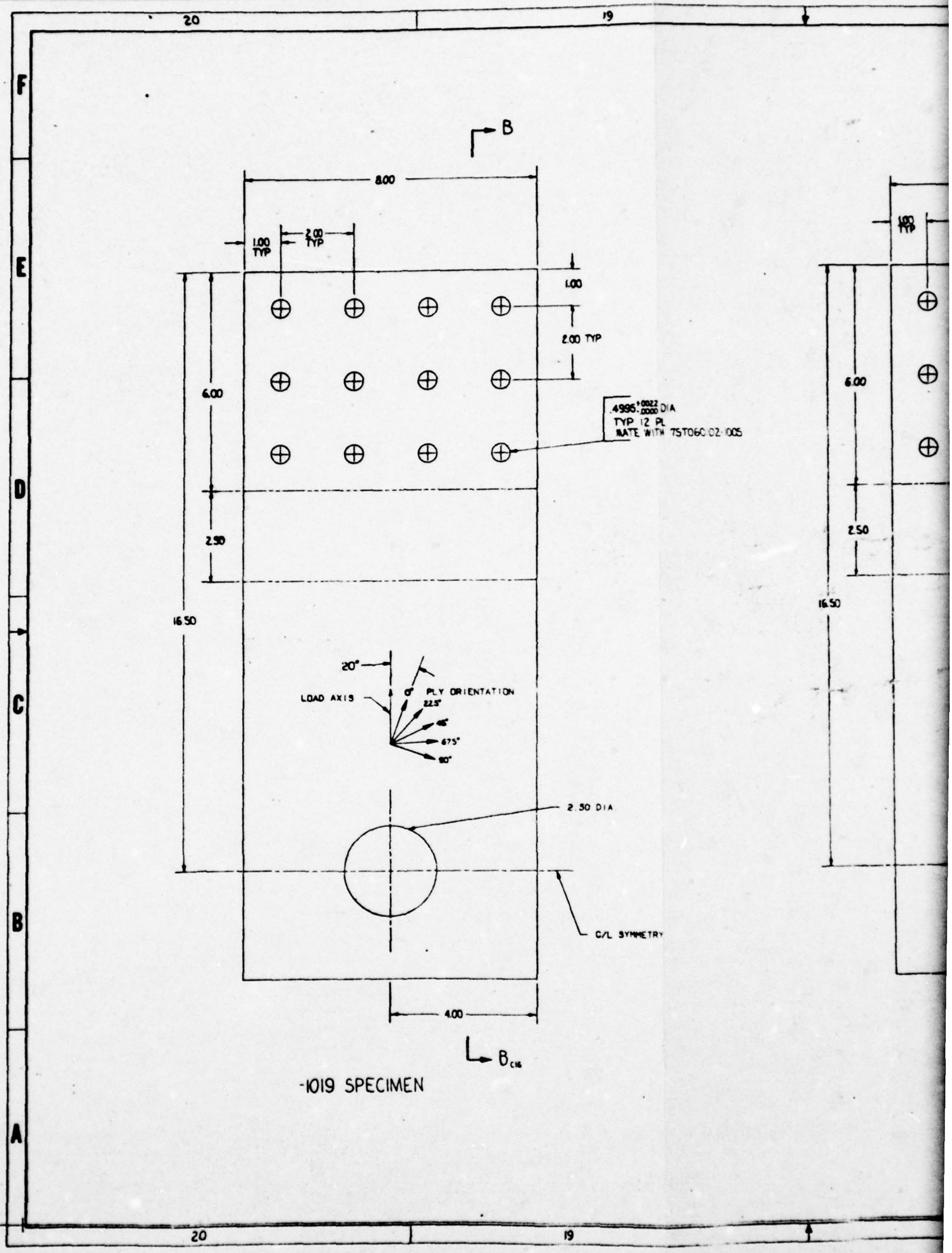
TOLERANCE UNLESS NOTED .XX ± .03 .XXX ± .010

MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION

SPECIMENS-COMPOSITE DAMAGE REPAIR TEST

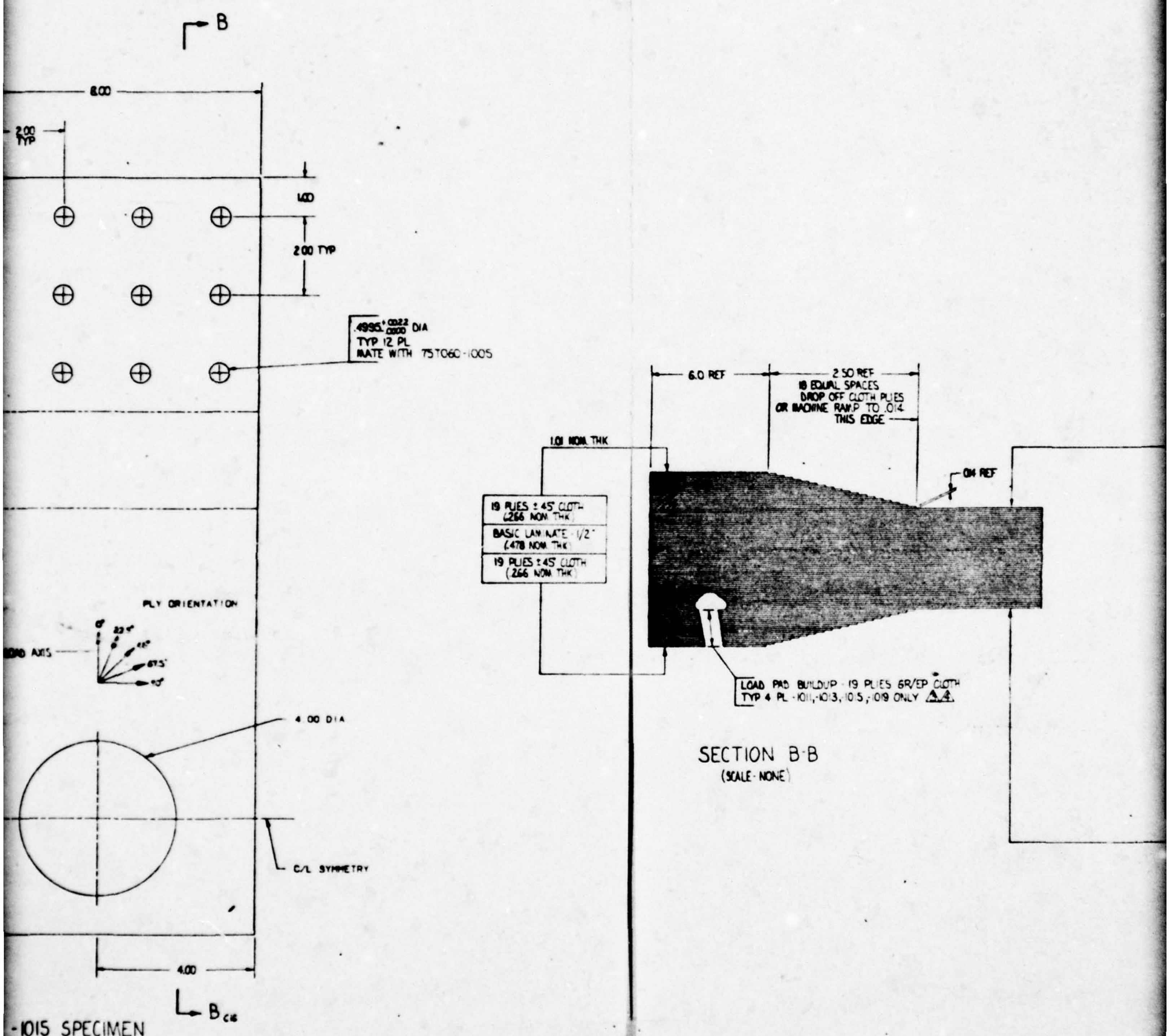
CODE IDENT NO 76301

75T060101



20
19
F
E
D
C
B
A
B
8.00
1.00 TYP
2.00 TYP
1.00
2.00 TYP
6.00
.4995 +.0022 .0000 DIA
TYP 12 PL
MATE WITH 7ST060102-1005
2.50
16.50
20°
LOAD AXIS
0° PLY ORIENTATION
22.5°
45°
67.5°
90°
2.50 DIA
C/L SYMMETRY
4.00
B C16
-1019 SPECIMEN

18 17 16



SECURITY CLASSIFICATION	DRAWING NUMBER	REV	SH
UNCLASSIFIED	75T060101		2

17 16

15 | 14 | 13

LTR	DESCRIPTION
A	SEE ECN A

PLY NO.	ORIEN-TATION	MATL	PLY NO.	ORIEN-TATION	MATL
1	45	△	27	-45	△
2	-45	△	28	45	△
3	0	△	29	0	△
4	45	△	30	-45	△
5	-45	△	31	45	△
6	0	△	32	0	△
7	0	△	33	-45	△
8	90	△	34	45	△
9	0	△	35	0	△
10	0	△	36	0	△
11	45	△	37	-45	△
12	-45	△	38	45	△
13	0	△	39	0	△
14	0	△	40	0	△
15	45	△	41	-45	△
16	-45	△	42	45	△
17	0	△	43	0	△
18	0	△	44	0	△
19	45	△	45	90	△
20	-45	△	46	0	△
21	0	△	47	0	△
22	45	△	48	-45	△
23	-45	△	49	45	△
24	0	△	50	0	△
25	45	△	51	-45	△
26	-45	△	52	45	△

1/2" BASIC LAMINATE

.478 NOM. THK.

FIGURE A-1 CONTINUED

A-3

SEE SHEET 1 FOR NOTES AND PARTS LIST

QTY REQD	QTY REQD	CODE IDENT NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE
	FIND NO			PARTS LIST		

DASH NO.	QTY REQD	NEXT ASSY	USED ON
-1019	1	75T060106	TEST
-1017	1		TEST
-1015	4	75T060106	TEST

TOLERANCE UNLESS NOTED

FINISH SPEC

CONTR NO

3-6-78

MCDONNELL

SPECIMENS

DAMAGE R

SIZE J | CODE IDENT NO 76301 | 75

SCALE FULL | WT

15 | 14 | 13

3

REV	DESCRIPTION	DATE	APPROVED
A	SEE DCN A		

ORIENTATION	MATL
-45	△1
45	△1
0	△1
-45	△1
45	△1
0	△1
-45	△2
45	△2
0	△1
0	△1
-45	△1
45	△1
0	△1
0	△1
-45	△1
45	△1
0	△1
0	△1
90	△1
0	△1
0	△1
-45	△2
45	△2
0	△1
-45	△2
45	△2

FIGURE A-1 CONTINUED

A-3

SEE SHEET 1 FOR NOTES AND PARTS LIST

QTY REQD	QTY REQD	CODE IDENT NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE	DRAWING OR SPECIFICATION NO	NOTE NO	ZONE

PARTS LIST

DASH NO.	QTY REQD (FIND NO / PHD ASSY)	NEXT ASSY	USED ON
-1019	1	75T060106	TEST
-1017	1		TEST
-1015	4 △4	75T060106	TEST

APPLICATION

TOLERANCE UNLESS NOTED
XX = ±.00
XXX = ±.00

FINISH SPEC

CONTR NO

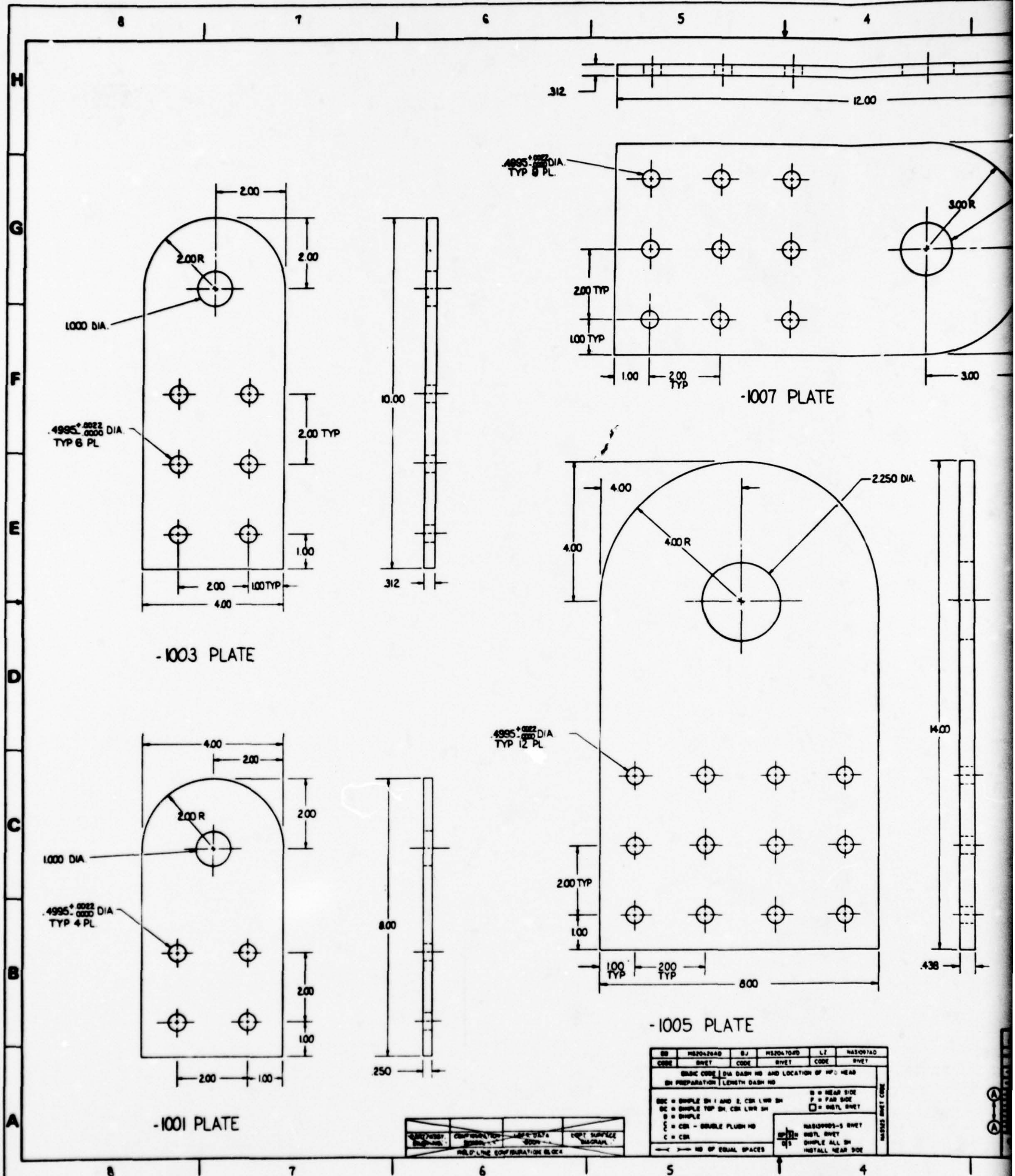
MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION

SPECIMENS-COMPOSITE DAMAGE REPAIR TEST

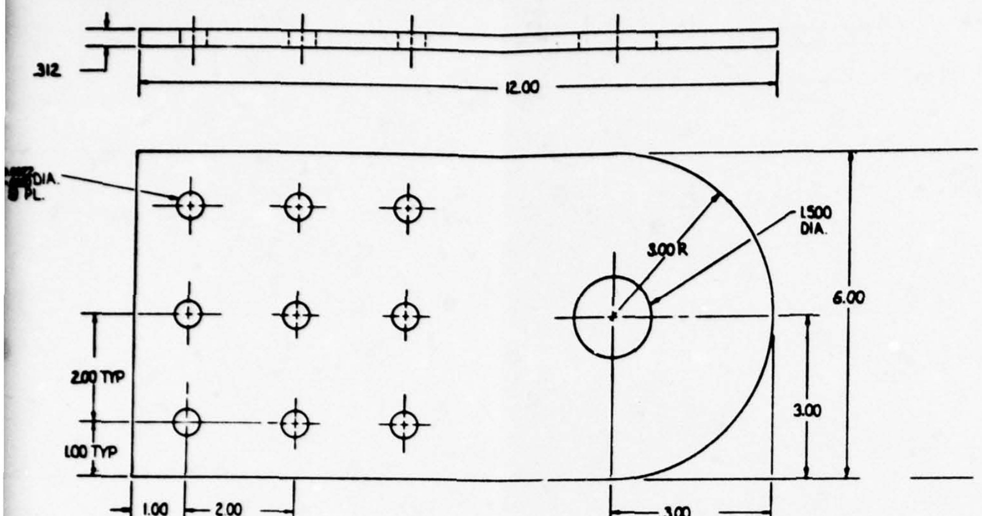
SIZE J | CODE IDENT NO 76301 | 75T060101

SCALE | ACTUAL WT | LB | SHEET 2 OF 2

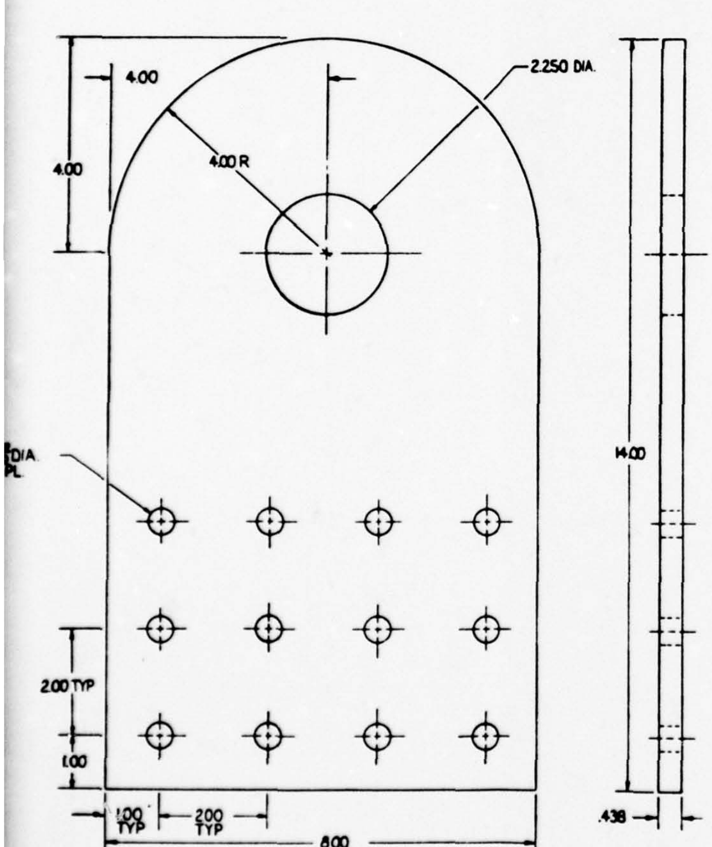
MAC 160FCCB (REV 30 SEP 70)



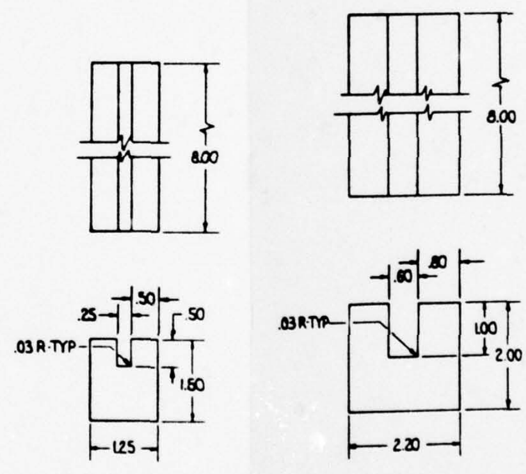
.312
12.00
.4995 DIA. TYP 9 PL.
3.00 R
2.00 TYP
1.00 TYP
1.00
2.00 TYP
3.00
-1007 PLATE
2.00
2.00 R
1.000 DIA.
.4995 +.0022 -.0000 DIA. TYP 6 PL
2.00 TYP
1.00
10.00
2.00
1.00 TYP
4.00
.312
-1003 PLATE
4.00
2.00 R
2.250 DIA.
4.00 R
.4995 DIA. TYP 12 PL
14.00
2.00 TYP
1.00
1.00 TYP
2.00 TYP
8.00
.438
-1005 PLATE
4.00
2.00
2.00 R
1.000 DIA.
.4995 +.0022 -.0000 DIA. TYP 4 PL
8.00
2.00
1.00
2.00
1.00
.250
-1001 PLATE



-1007 PLATE



-1005 PLATE



-1009 BLOCK

-1011 BLOCK

FIGURE A-2 DETAILED DRAWING OF LOAD PLATES FOR COMPOSITE REPAIR LAMINATES

A-4

4 HEAT TREAT TO 150-180 KSI, PER P.S. 15014
3 MARK PARTS PER P.S. 16001
2 MIN. SURFACE FINISH - 125/. TYP ALL SURFACES.
NOTES: 1. BREAK ALL SHARP CORNERS TO .01-.03 R

QTY REQD	QTY REQD	FSCM NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STF NO	STOCK	MATERIAL OR MATERIAL CODE	DRAWING OR SPECIFICATION NO	NOTE NO	ZONE
			-1011	BLOCK		2.00×2.20×8.00	4130 STEEL	MIL-S-18729 COND A	4	C1
			-1009	BLOCK		1.25×1.50×8.00	4130 STEEL	MIL-S-18729 COND A	4	C3
			-1007	PLATE		.312×6.00×12.00	4130 STEEL	MIL-S-18729 COND N	4	F5
			-1005	PLATE		.500×8.00×14.00	4130 STEEL	MIL-S-18729 COND N	4	B5
			-1003	PLATE		.312×4.00×8.00	4130 STEEL	MIL-S-18729 COND N	4	D8
			-1001	PLATE		.250×4.00×8.00	7075-T651 ALUM.	QQ-A-250/12		A8

PARTS LIST

DASH NO	QTY REQD	NEXT ASSY	USED ON
-1011	2	75T060105	TEST
-1009	2	75T060105	TEST
-1007	4	75T060106	TEST
-1005	4	75T060105	TEST
-1003	4	75T060105	TEST
-1001	4	75T060105	TEST

APPLICATION

TOLERANCE UNLESS NOTED
.XX = ±.03
.XXX = ±.010

MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION

PLATES, LOAD
COMPOSITE REPAIR TEST

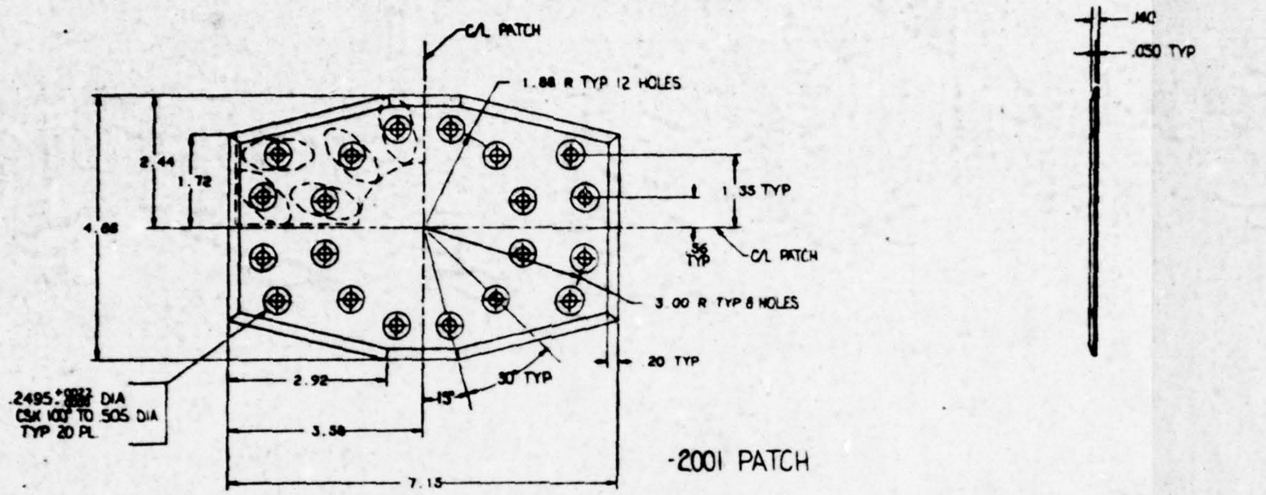
SIZE E | FSCM NO 76301 | DWG NO 75T060102

SCALE / FULL | SHEET 1 OF 1

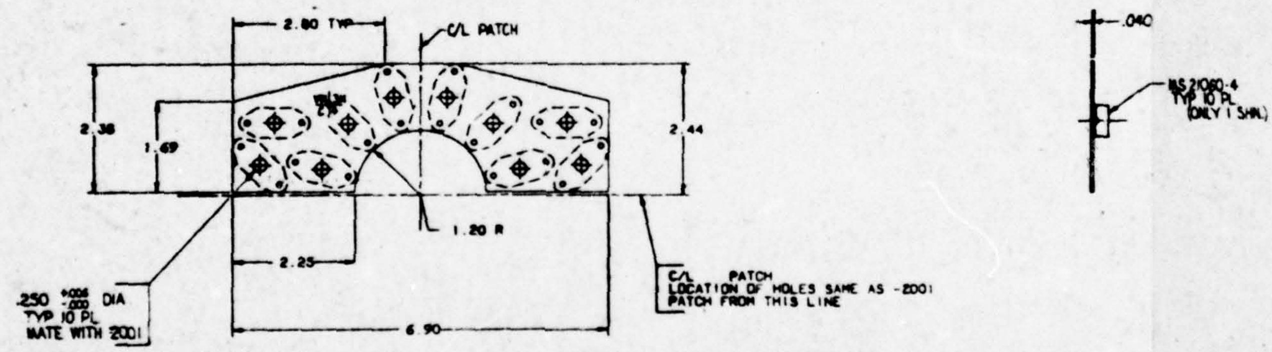
J. Watson 3-6-78

UNCLASSIFIED

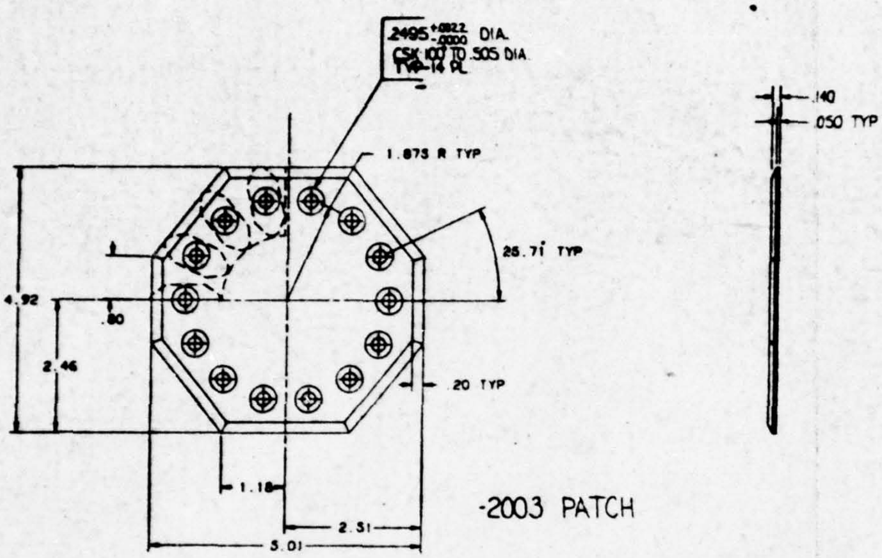
2



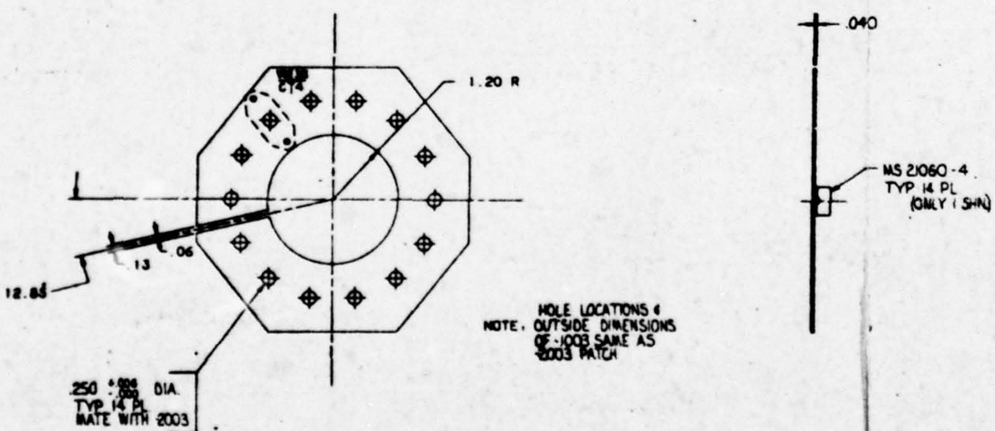
-2001 PATCH



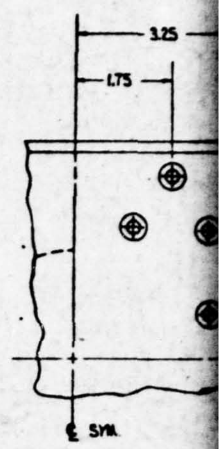
-1001 BACKING PLATE



-2003 PATCH

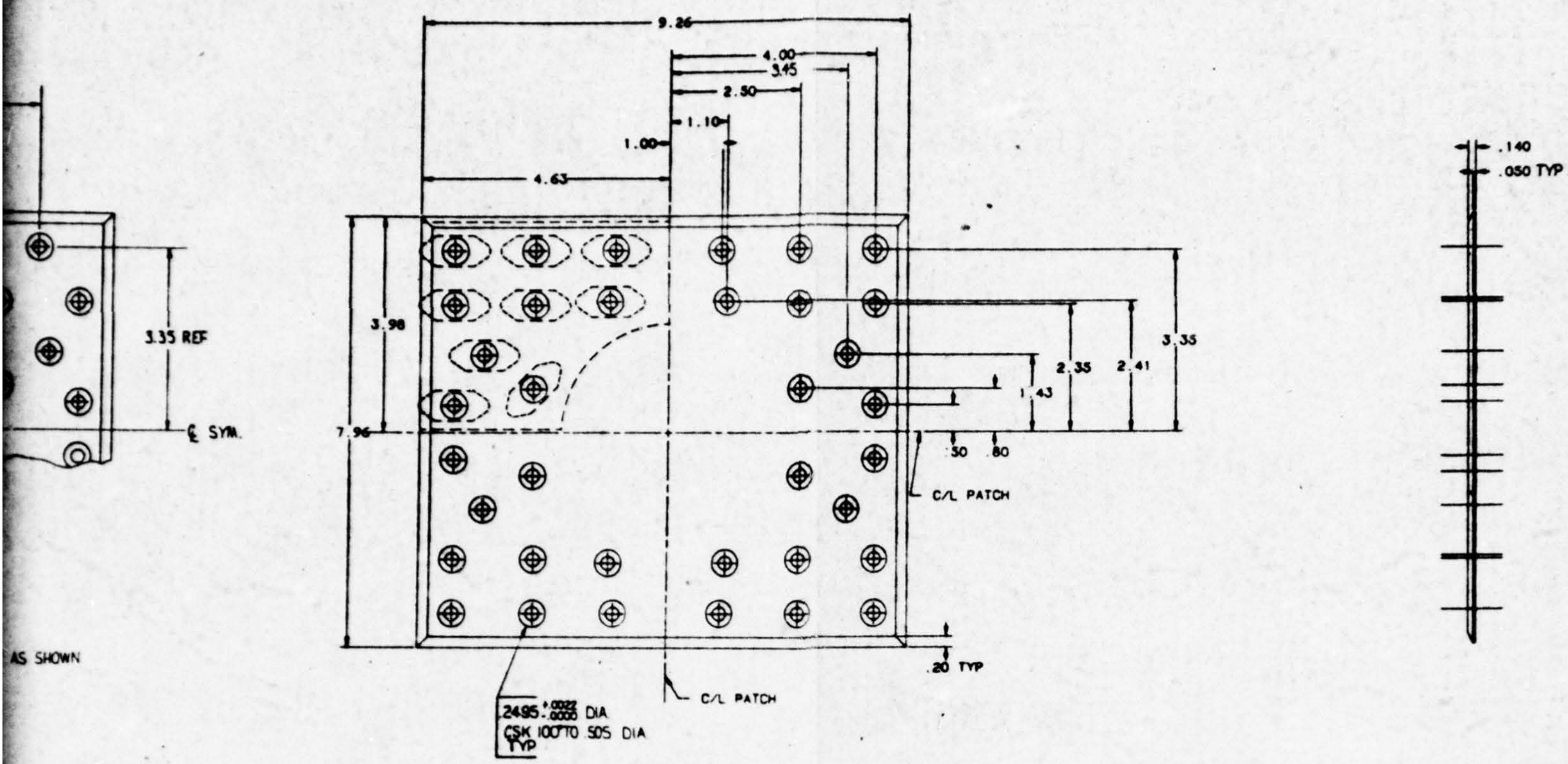


-1003 BACKING PLATE

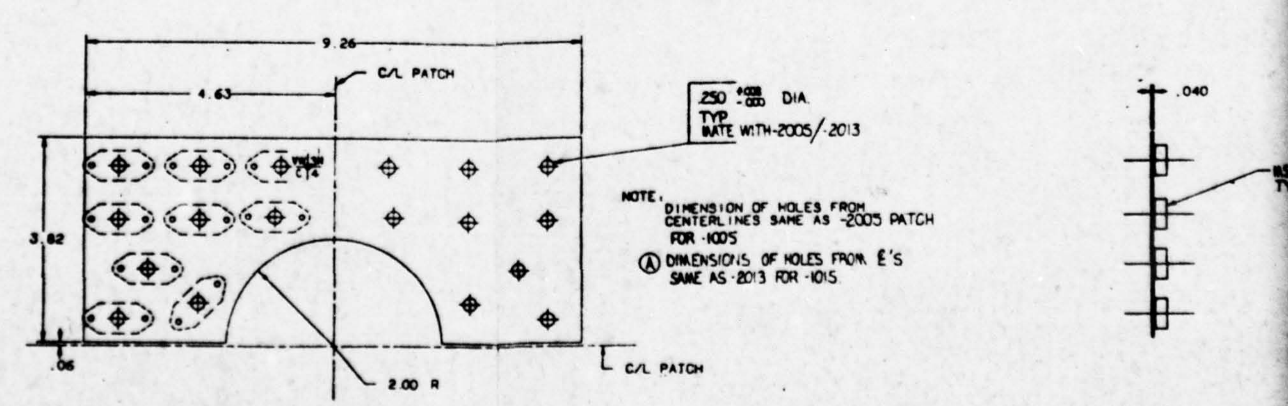


Ⓐ -2013 PATCH
SAME AS -2005 EXCEPT

UNCLASSIFIED | 75T060103 | A



-2005 PATCH



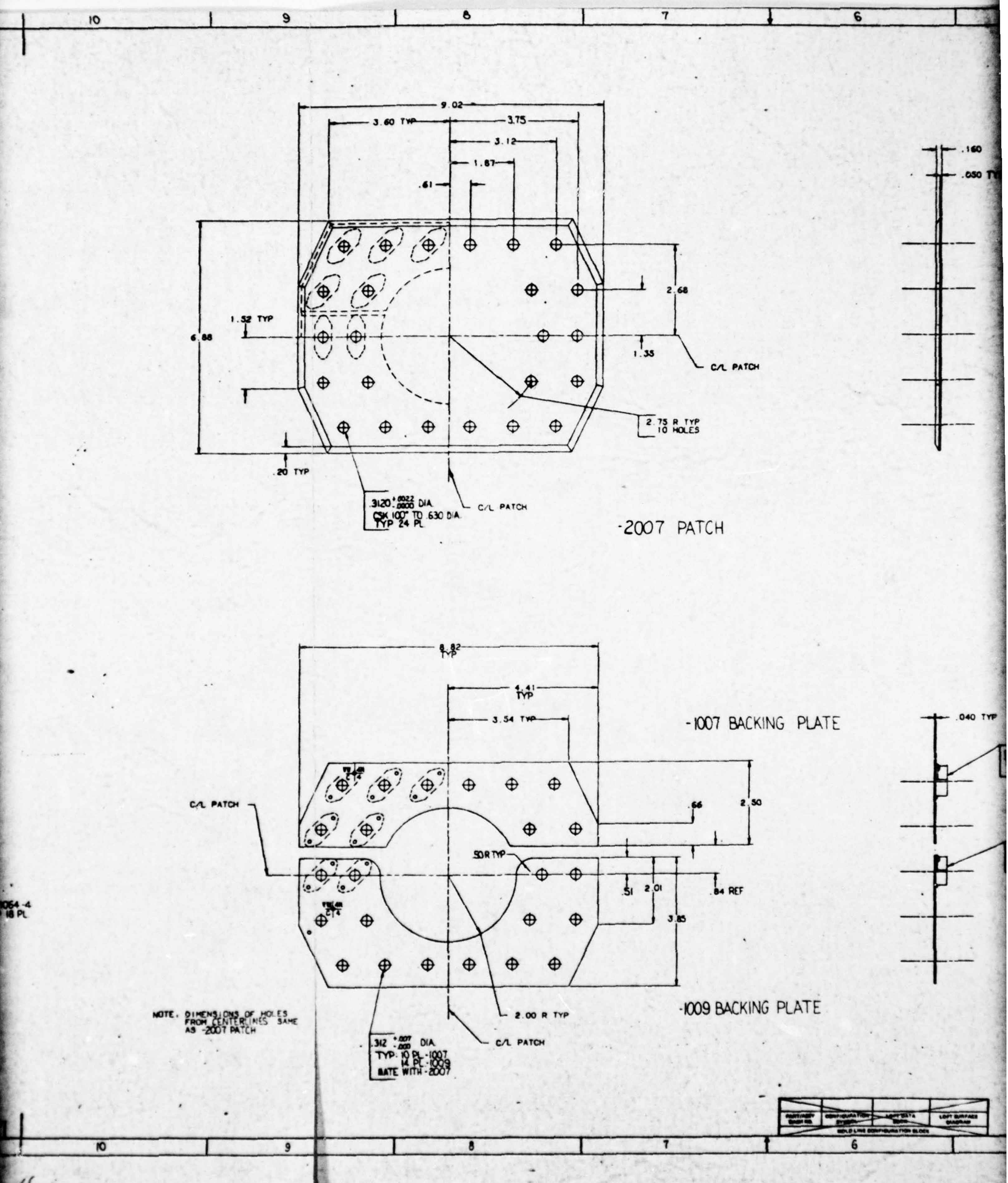
NOTE:
DIMENSION OF HOLES FROM CENTERLINES SAME AS -2005 PATCH FOR -1005
Ⓐ DIMENSIONS OF HOLES FROM ℄'S SAME AS -2013 FOR -1015

-1005 BACKING PLATE (SHOWN)
-1015 BACKING PLATE Ⓐ

UNCLASSIFIED 75T060103

15 14 13 12 11

3

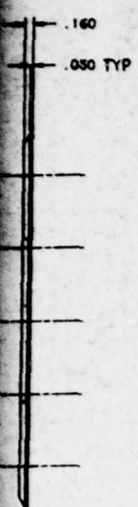

9.02
3.60 TYP
3.75
3.12
1.87
.61
2.68
1.52 TYP
6.88
1.35
C/L PATCH
2.75 R TYP
10 HOLES
.20 TYP
.3120 +.0022 -.0000 DIA.
CSK 100° TO .630 DIA.
TYP 24 PL
C/L PATCH
-2007 PATCH
.160
.050 TYP
8.82
TYP
4.41
TYP
3.54 TYP
-1007 BACKING PLATE
.040 TYP
C/L PATCH
.66
2.50
.50 R TYP
.51
2.01
.84 REF
3.85
2.00 R TYP
C/L PATCH
NOTE: DIMENSIONS OF HOLES FROM CENTERLINES SAME AS -2007 PATCH
.312 +.007 -.000 DIA
TYP: 10 PL-1007
14 PL-1009
MATE WITH -2007
-1009 BACKING PLATE
1064-4
18 PL
10
9
8
7
6

5	4	3	2	1

THIRD ANGLE PROJECTION

REVISIONS

ZONE	REV	DESCRIPTION	DATE	APPROVED
	A	ADDED -2015 & -1015	16 JULY 78	



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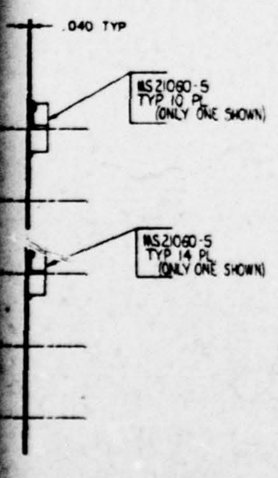
3. MARK PARTS PER P.S. 16001.
2. MIN. SURFACE FINISH - 125 TYP ALL SURFACES
NOTES: 1. BREAK ALL SHARP CORNERS TO .01-.03 R.

FIGURE A-3. DETAILED DRAWING OF REPAIR PATCHES FOR 3/16 LAMINATES



PARTS LIST

QTY	QTY	QTY	QTY	QTY	QTY	QTY REQD	QTY REQD	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE	DRAWING OR SPECIFICATION NO	NOTE NO	ZONE
								-2015	PATCH	.140×8.00×9.30	6AL-4V TITANIUM	MIL-T-9046 TYPE III COMP C COND ANNEALED	1,2,3	F16
								-2011	PATCH	.160×2.90×3.70	6AL-4V TITANIUM	MIL-T-9046 TYPE III COMP C COND ANNEALED	1,2,3	
								-2009		.140×3.50×4.00				
								-2007		.160×7.00×9.10				E7
								-2005		.140×8.00×9.30				B14
								-2003		.140×5.00×5.00				E18
								-2001	PATCH	.140×5.00×7.20				E23
								-1015	BACKING PLATE	.040×4.00×9.30				A14
								-1013	BACKING PLATE	.040×.90×2.70				
								-1011		.040×1.75×3.50				
								-1009		.040×3.90×8.90				B7
								-1007		.040×2.50×8.90				B7
								-1005		.040×4.00×9.30				A14
								-1003		.040×5.00×5.00				B19
								-1001	BACKING PLATE	.040×2.40×6.90	6AL-4V TITANIUM	MIL-T-9046 TYPE III COMP C COND ANNEALED	1,2,3	B24
	4		28	20				3M391-4-4	RIVET					
32		6			36	28	20	3M391-3-4	RIVET					
	2		14	10				MS21060-5	PLATE NUT					
16		3			18	14	10	MS21060-4	PLATE NUT					

CONT'D ON SH 2

DASH NO.	QTY REQD	NEXT ASSY	USED ON
-1013	2	75T060103	TEST
-1011	2		TEST
-1009	1		TEST
-1007	1		TEST
-1005	2		TEST
-1003	1		TEST
-1001	2	75T060103	TEST

MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION

REPAIR PATCHES
3/16 PANELS - COMPOSITE DAMAGE REPAIR TEST

J. C. Watson 8-20-78

SIZE J | CODE 76301 | DWG NO. 75T060103

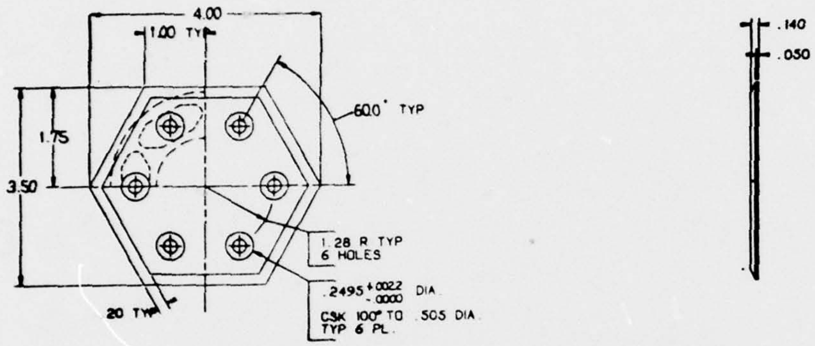
UNCLASSIFIED | 75T060103

5

F

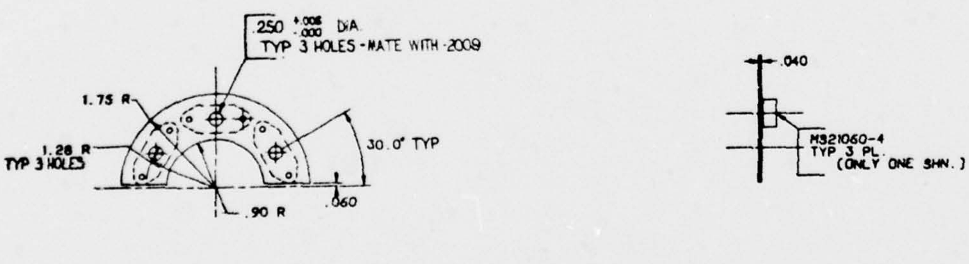
E

D



-2009 PATCH

C

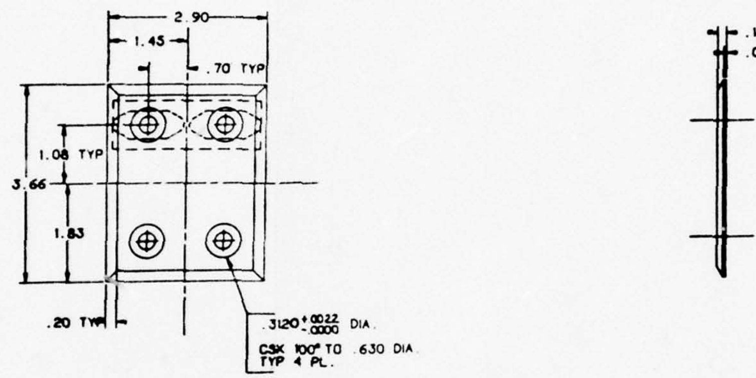


B

-1011 BACKING PLATE

A

SECURITY CLASSIFICATION	DRAWING NUMBER	REV	SH
UNCLASSIFIED	75T060103		2



-2011 PATCH

-1013 BACKING PLATE

FIGURE A-3 CONTINUED

SEE SHEET 1 FOR PARTS LIST AND NOTES

DASH NO.	QTY REQD	CODE IDENT NO	PART OR IDENTIFYING NO
-2011	2	75T060105	TEST
-2009	2		
-2007	1		
-2005	1		
-2003	1		
-2001	1	75T060105	TEST
	FIN. ART	NEXT ASSY	USED ON
	QTY REQD	APPLICATION	

PARTS LIST

CONT'D FROM SH 1

TOLERANCE UNLESS NOTED
XX = ±.03
XXX = ±.010

DWN B. GIRLSER
STRF
MIP
J.C. Watson 3-20-7

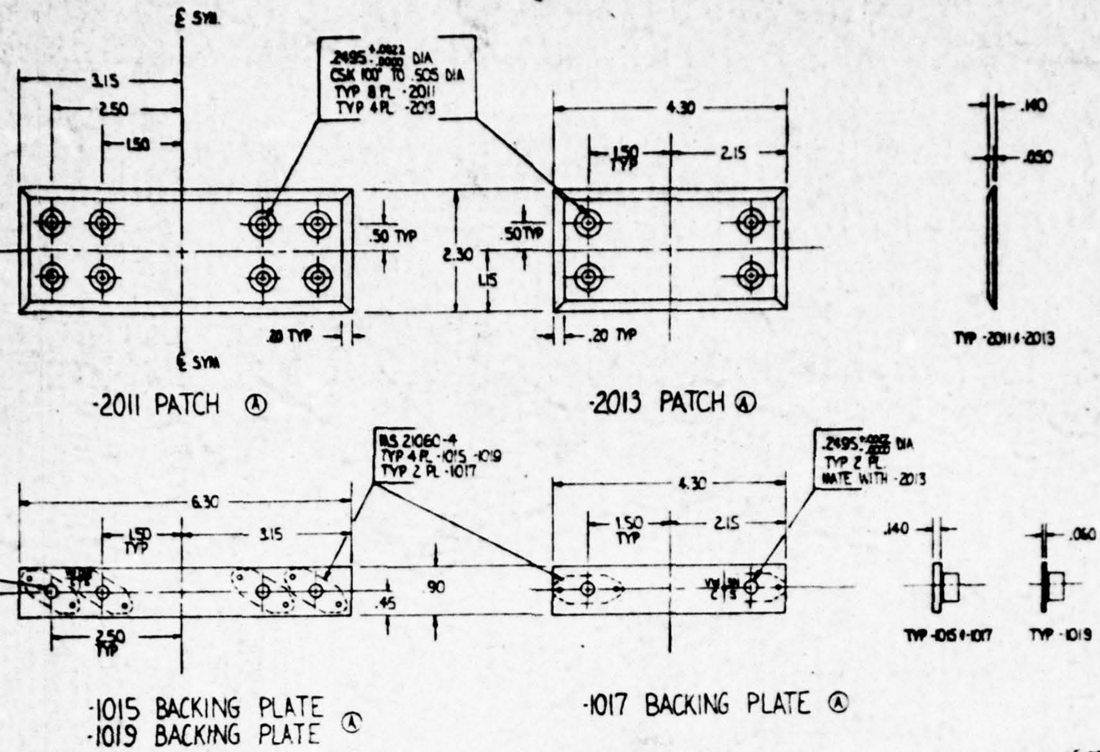
MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION

REPAIR PATCHES
3/16" PANELS - COMPOSITE
DAMAGE REPAIR TEST

SIZE J | CODE IDENT NO. 76301 | 75T060103

SCALE/FULL | WT | LB | SHEET 2 OF 2

MAC 1607CCP (REV 30 SEP 70)



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NOTES: 1. BREAK ALL SHARP CORNERS .01-.03 R
2. MIN SURFACE FINISH 125 TYP ALL SURFACES
3. MARK PARTS PER PS 16001

FIGURE A-4 DETAILED DRAWING OF REPAIR PATCHES FOR 1/2 LAMINATES

A-7

Qty reqd	Part or identifying no.	Nomenclature or description	Stock	Material or material code	Notes	Zone
	-2013	PATCH	.140×2.30×4.30	6AL-4V TITANIUM	1,2,3	
	-2011	PATCH	.140×2.30×6.30			
	-1019	BACKING PLATE	.060×.90×6.30			
	-1017	BACKING PLATE	.140×.90×4.30			
	-1015	BACKING PLATE	.140×.00×6.30			
	-2007	PATCH	.140×8.20×10.70	6AL-4V TITANIUM	1,2,3	
	-2005		.160×8.00×11.00			
	-2003		.160×5.00×8.90			
	-2001	PATCH	.140×4.80×6.80	6AL-4V TITANIUM	1,2,3	
	-2009	PATCH	.160×4.80×6.80	301 CRES	1,2,3	
28, 14, 4	3M391-4-5	RIVET				
8, 4, 8, 36, 14, 7, 16, 6	3M391-3-5	RIVET				
14, 7, 2	MS21060-5	PLATE NUT				
4, 2, 4, 18, 8, 3	MS21060-4	PLATE NUT				
	-1013	BACKING PLATE	.160×2.40×6.80	301 CRES	1,2,3	C24
	-1011	BACKING PLATE	.140×4.10×10.40	6AL-4V TITANIUM	1,2,3	B7
	-1009		.140×3.90×10.90			B12
	-1007		.140×3.20×8.00			C18
	-1005		.140×2.40×6.80			C24
	-1003		.140×.90×2.70			A18
	-1001	BACKING PLATE	.140×1.80×3.60	6AL-4V TITANIUM	1,2,3	A24

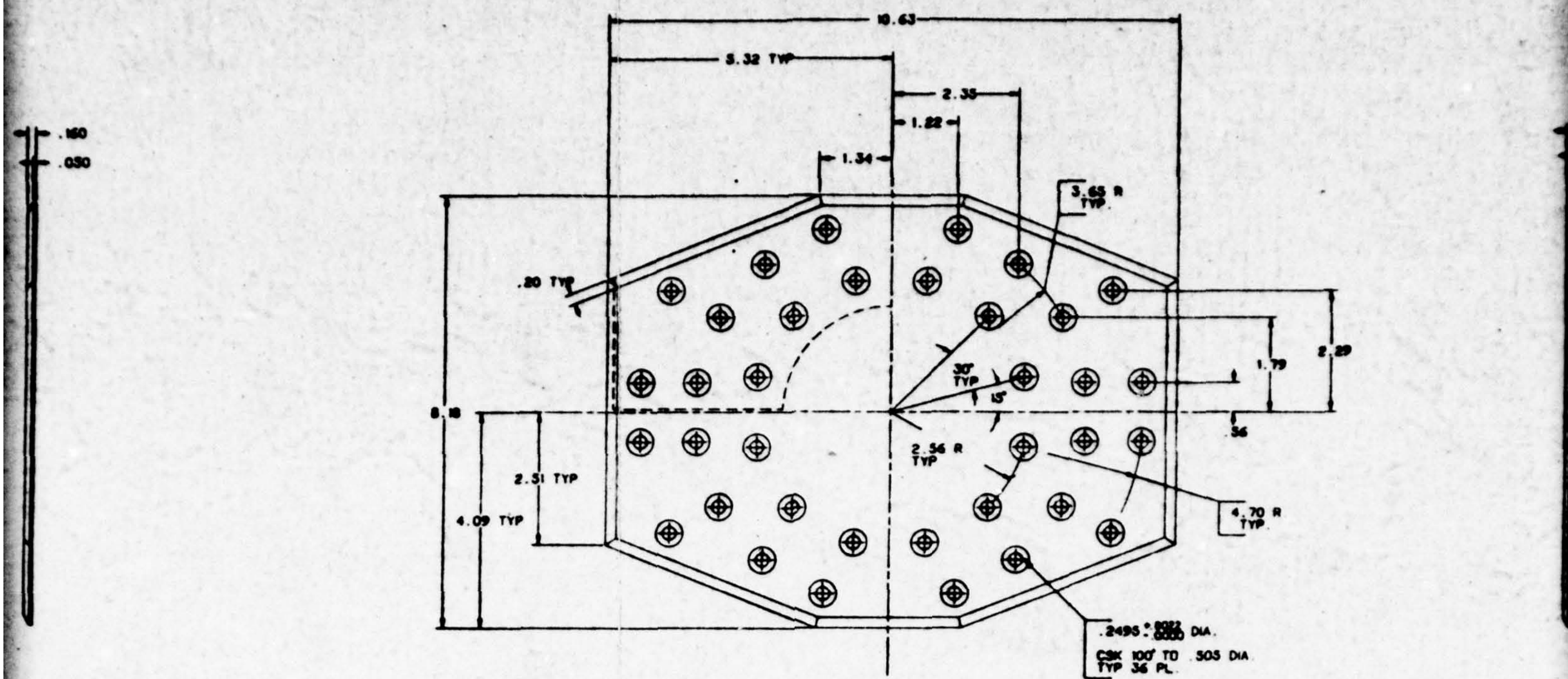
Dash no.	Qty reqd	Next assy	Used on
-1019	4	75T060105	TEST
-1013	4	75T060104	TEST
1015-1017 -2009	2	75T060106	TEST
-1011	6	75T060105	TEST
-2011 & -2007	3	75T060105	TEST
2013 & -2001 TO -2005	1 EACH	75T060106	TEST
-1001 TO -1009	2 EACH	75T060106	TEST

McDONNELL AIRCRAFT COMPANY
McDONNELL DOUGLAS CORPORATION

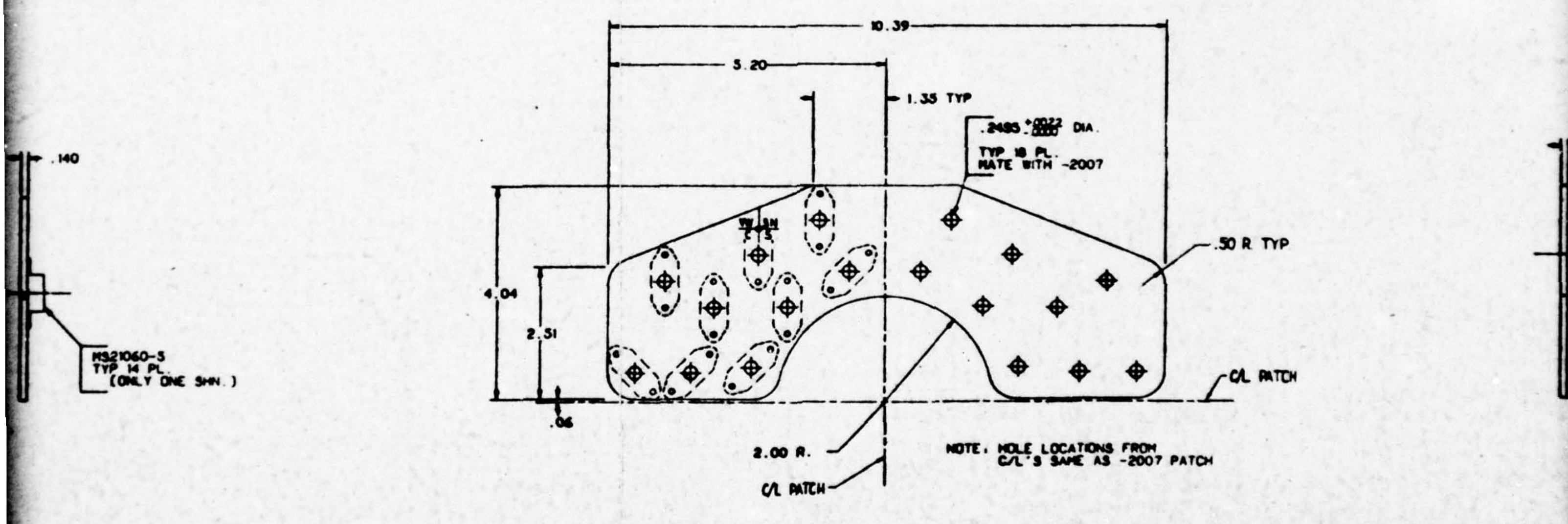
REPAIR PATCHES
1/2" PANELS - COMPOSITE
DAMAGE REPAIR TEST

76301 75T060104

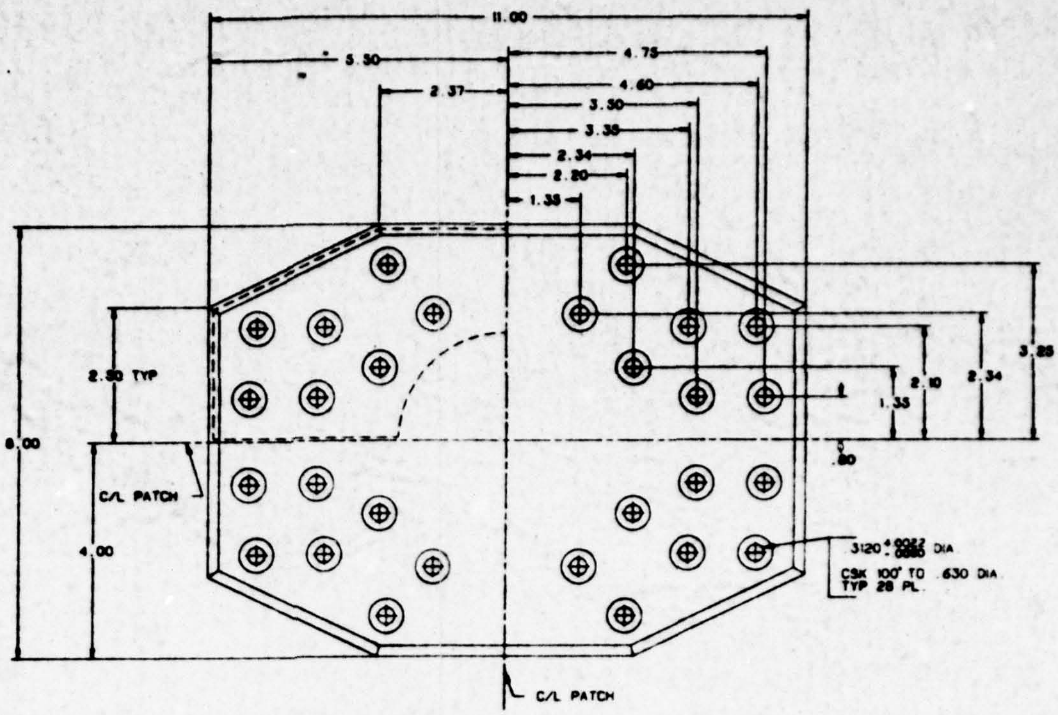
UNCLASSIFIED 75T060104



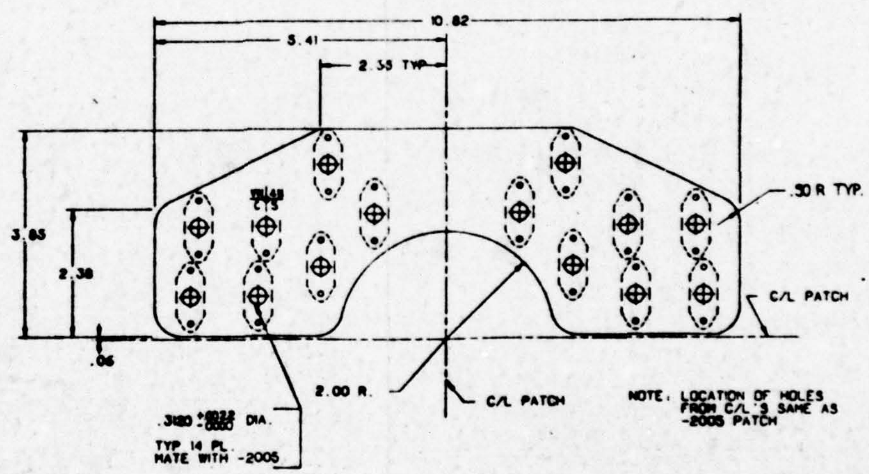
-2007 PATCH



-1011 BACKING PLATE

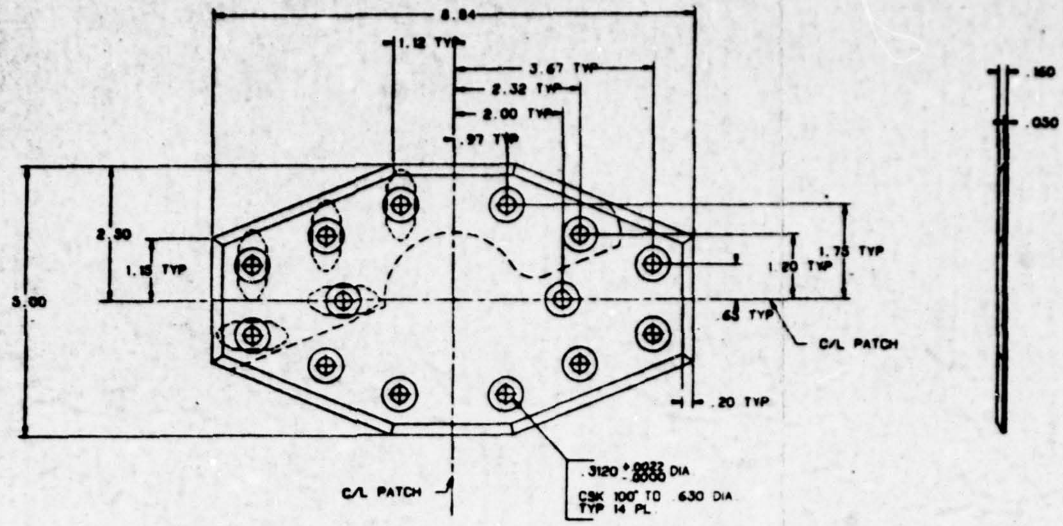


-2005 PATCH

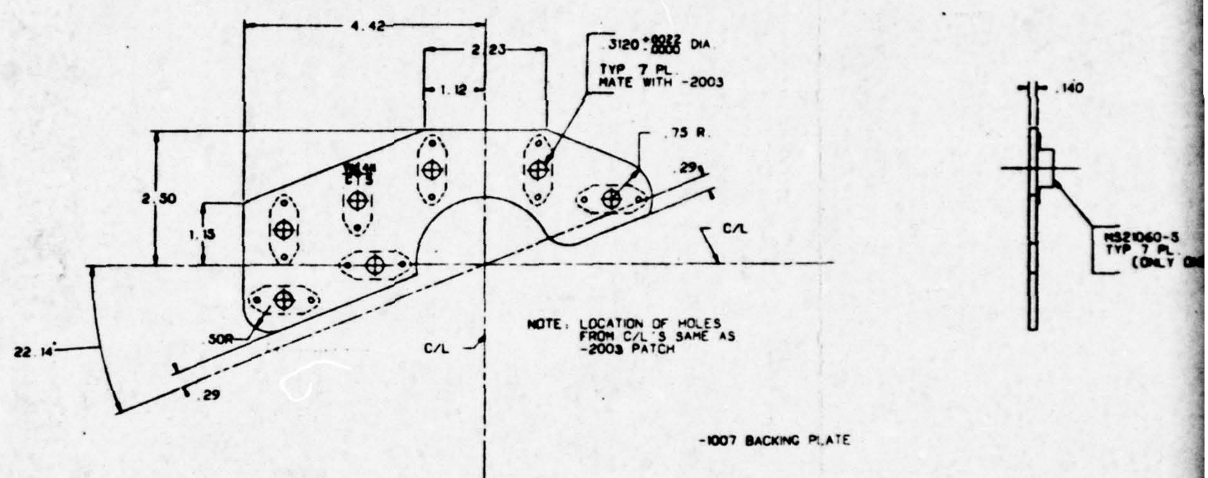


-1809 BACKING PLATE

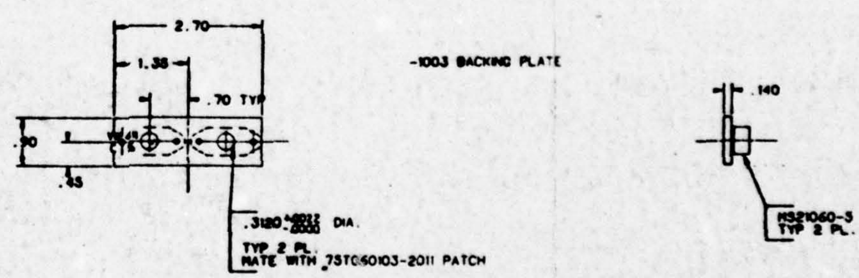
UNCLASSIFIED | 75T060



-2003 PATCH



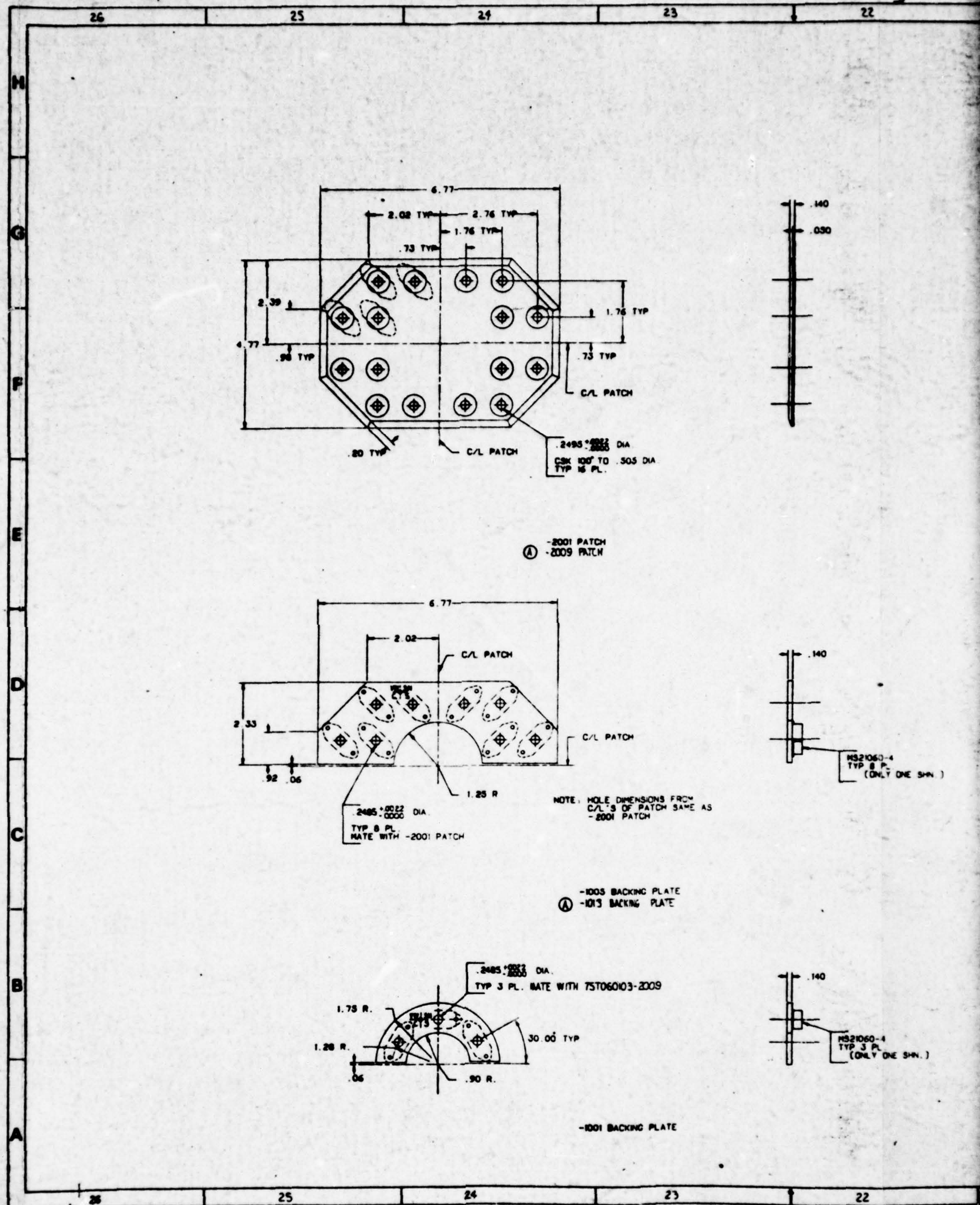
-1007 BACKING PLATE



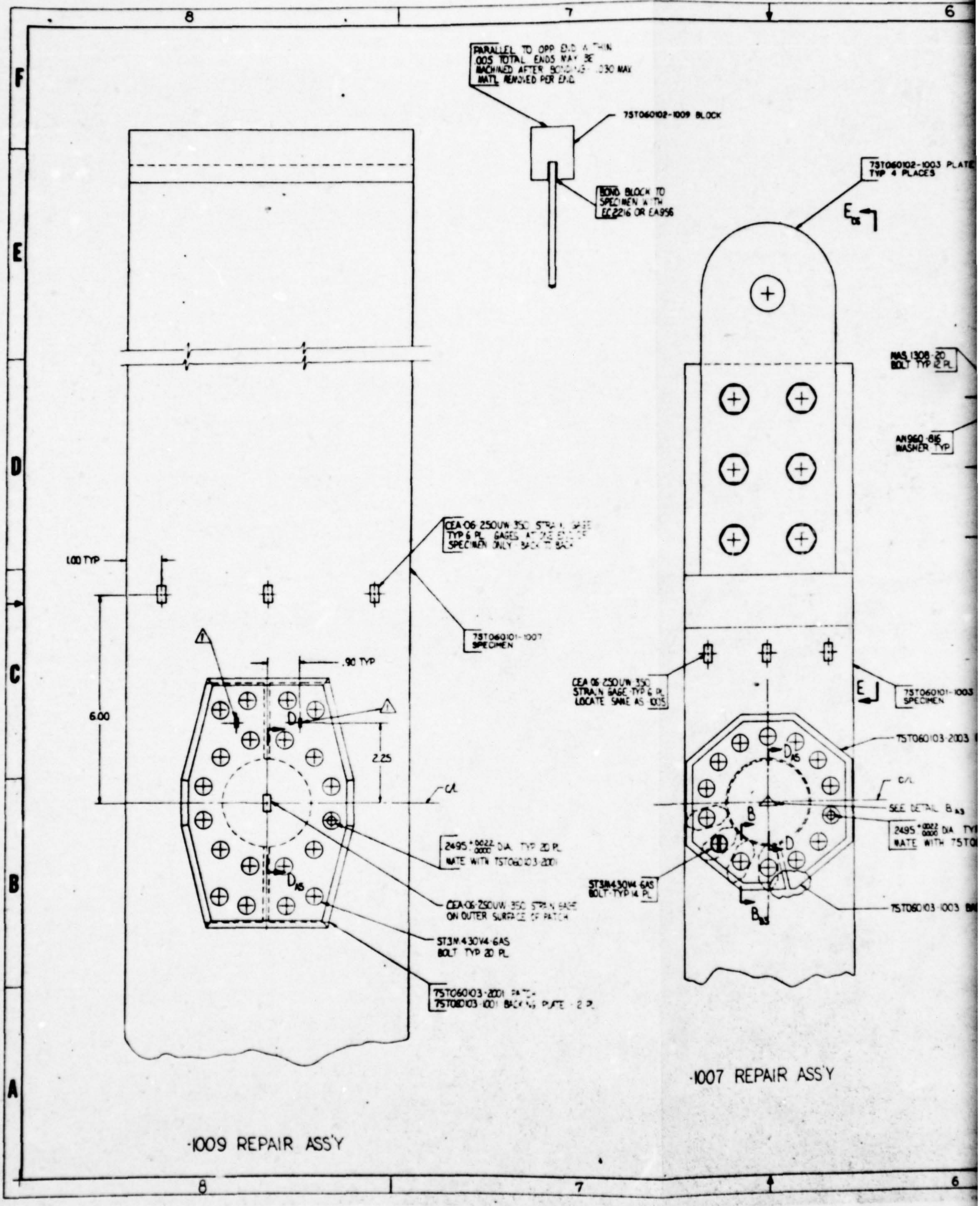
-1003 BACKING PLATE

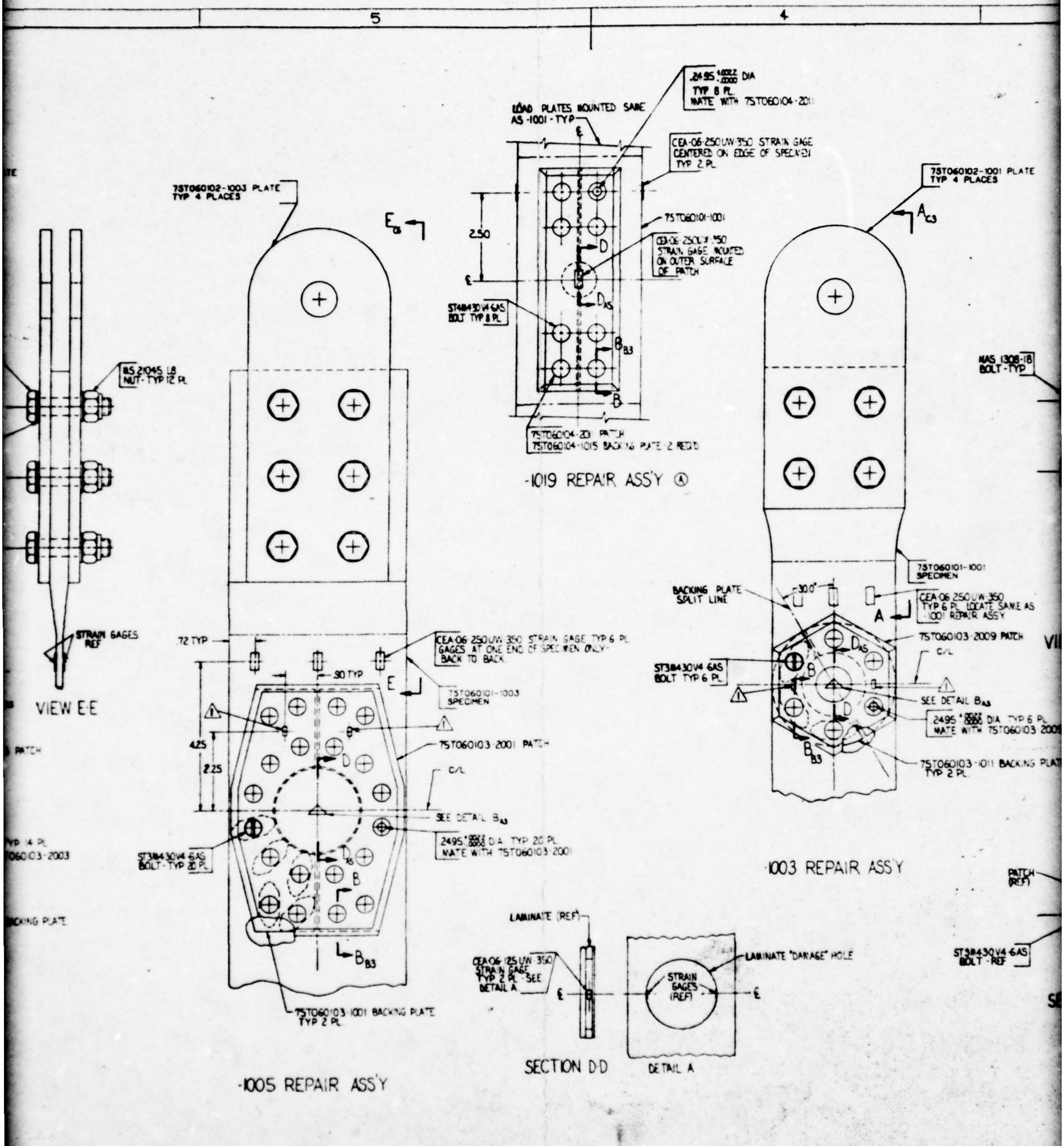
UNCLASSIFIED 75T060104 1 A

21	20	19	18	17



26
25
24
23
22
H
G
F
E
D
C
B
A
6.77
2.02 TYP
2.76 TYP
1.76 TYP
.73 TYP
2.39
4.77
.98 TYP
1.76 TYP
.73 TYP
C/L PATCH
.20 TYP
.2495 +.0022 -.0000 DIA.
CSK 100° TO .505 DIA.
TYP 16 PL.
.140
.030
-2001 PATCH
Ⓐ -2009 PATCH
6.77
2.02
C/L PATCH
2.33
C/L PATCH
.92
.06
1.25 R
.2485 +.0022 -.0000 DIA.
TYP 8 PL.
MATE WITH -2001 PATCH
NOTE: HOLE DIMENSIONS FROM C/L'S OF PATCH SAME AS -2001 PATCH
.140
MS21060-4
TYP 8 PL.
(ONLY ONE SHN.)
-1005 BACKING PLATE
Ⓐ -1013 BACKING PLATE
.2485 +.0022 -.0000 DIA.
TYP 3 PL. MATE WITH 75T060103-2009
1.75 R.
1.28 R.
30.00 TYP
.06
.90 R.
.140
MS21060-4
TYP 3 PL.
(ONLY ONE SHN.)
-1001 BACKING PLATE


PARALLEL TO OPP END WITHIN .005 TOTAL. ENDS MAY BE MACHINED AFTER BONDING .030 MAX MATL REMOVED PER END.
7ST060102-1009 BLOCK
BOND BLOCK TO SPECIMEN WITH EC2216 OR EA956
7ST060102-1003 PLATE TYP 4 PLACES
NAS 1308-20 BOLT TYP 12 PL
AN960-816 WASHER TYP
CEA-06-250UW-350 STRAIN GAGE TYP 6 PL. GAGES AT END OF SPECIMEN ONLY - BACK TO BACK
1.00 TYP
7ST060101-1007 SPECIMEN
.90 TYP
6.00
2.25
C/L
2495 DIA TYP 20 PL MATE WITH 7ST060103-2001
CEA-06-250UW-350 STRAIN GAGE ON OUTER SURFACE OF PATCH
ST3M430V4-6AS BOLT TYP 20 PL
7ST060103-2001 PATCH
7ST060103-1001 BACKING PLATE - 2 PL
CEA-06-250UW-350 STRAIN GAGE TYP 6 PL LOCATE SAME AS 1005
7ST060101-1003 SPECIMEN
7ST060103-2003
SEE DETAIL B
ST3M430V4-6AS BOLT TYP 14 PL
7ST060103-1003
-1007 REPAIR ASS'Y
-1009 REPAIR ASS'Y



SECURITY CLASSIFICATION	DRAWING NUMBER	REV	SH
UNCLASSIFIED	75T060105	A	1

2

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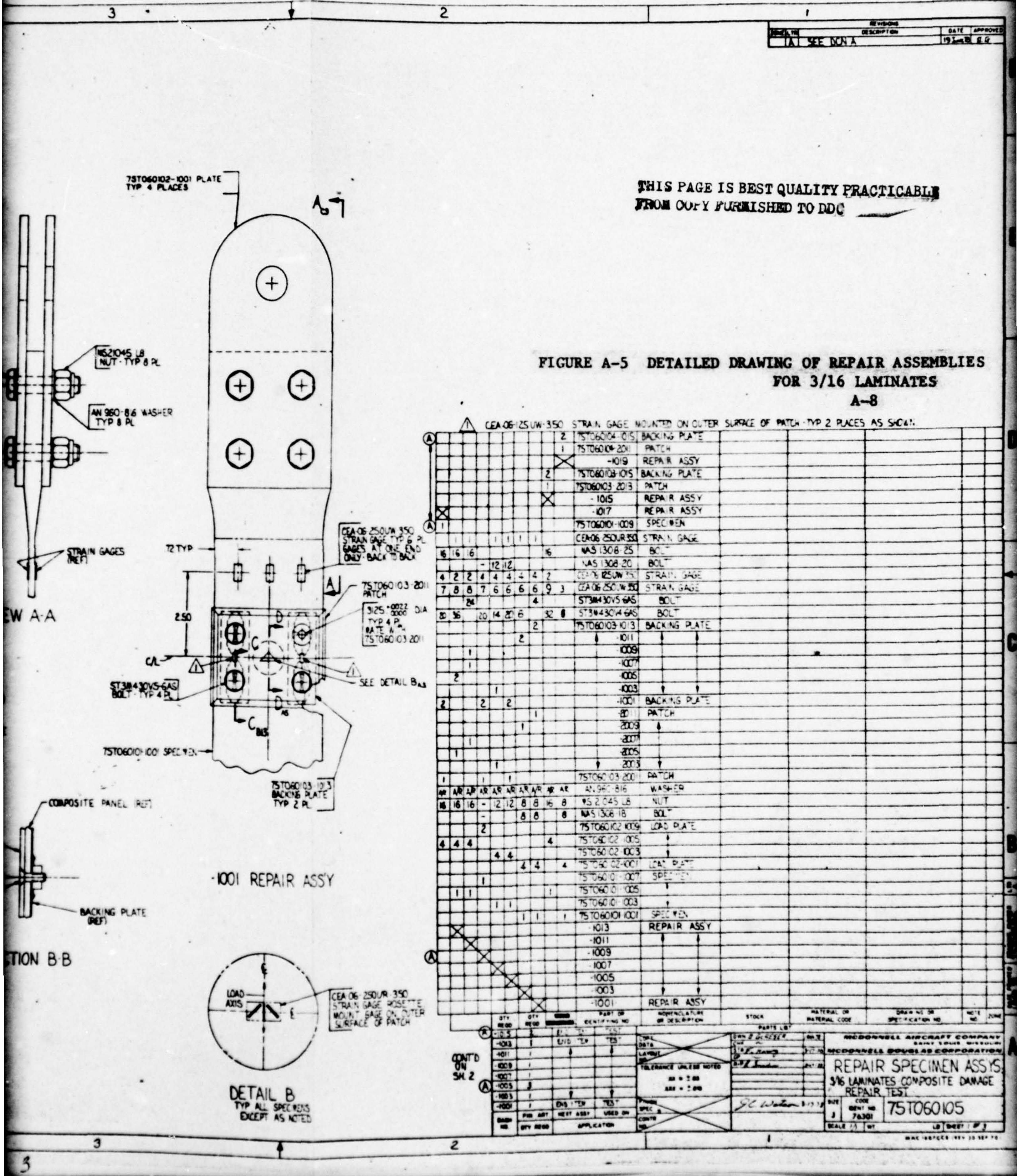
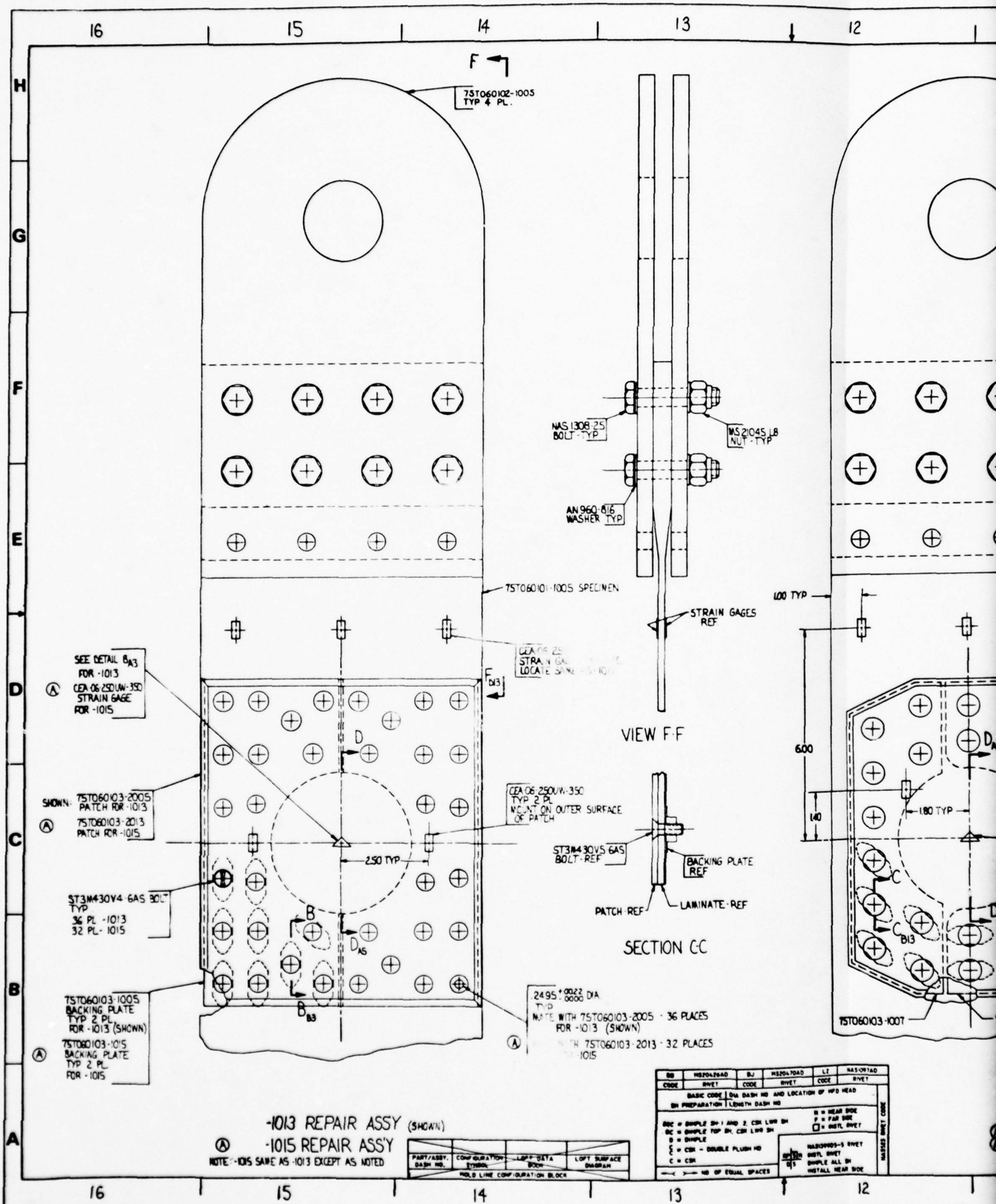


FIGURE A-5 DETAILED DRAWING OF REPAIR ASSEMBLIES FOR 3/16 LAMINATES



16
15
14
13
12
H
G
F
E
D
C
B
A
F
7ST060102-1005
TYP 4 PL.
NAS 1308-25
BOLT-TYP
MS 21045 L8
NUT-TYP
AN 960-816
WASHER TYP
7ST060101-1005 SPECIMEN
STRAIN GAGES
REF
1.00 TYP
SEE DETAIL B_{A3}
FOR -1013
CEA-06-250UW-350
STRAIN GAGE
FOR -1015
F_{D13}
VIEW F-F
D
D_{AS}
6.00
1.40
1.80 TYP
SHOWN: 7ST060103-2005
PATCH FOR -1013
7ST060103-2013
PATCH FOR -1015
CEA-06-250UW-350
TYP 2 PL
MOUNT ON OUTER SURFACE
OF PATCH
2.50 TYP
ST3M430VS-6AS
BOLT-REF
BACKING PLATE
REF
ST3M430V4-6AS BOLT
TYP
36 PL -1013
32 PL- 1015
PATCH REF
LAMINATE-REF
SECTION C-C
C
C_{B13}
B
B_{B3}
D_{AS}
7ST060103-1005
BACKING PLATE
TYP 2 PL
FOR -1013 (SHOWN)
7ST060103-1015
BACKING PLATE
TYP 2 PL
FOR -1015
.2495 +.0022 / -.0000 DIA
TYP
MATE WITH 7ST060103-2005 - 36 PLACES
FOR -1013 (SHOWN)
WITH 7ST060103-2013 - 32 PLACES
1015
7ST060103-1007
-1013 REPAIR ASSY (SHOWN)
-1015 REPAIR ASS'Y
NOTE: -1015 SAME AS -1013 EXCEPT AS NOTED
PART/ASSY. DASH NO.
CONFIGURATION SYMBOL
LOFT DATA BOOK
LOFT SURFACE DIAGRAM
MOLD LINE CONFIGURATION BLOCK
MS20426AD
BJ
MS20470AD
LZ
NAS1097AD
CODE
RIVET
BASIC CODE
DIA DASH NO AND LOCATION OF MFD HEAD
SH PREPARATION
LENGTH DASH NO
N = NEAR SIDE
F = FAR SIDE
INSTL RIVET
D = DIMPLE
C = CSK
NO OF EQUAL SPACES
NAS1909-5 RIVET
INSTL RIVET
DIMPLE ALL SH
INSTALL NEAR SIDE

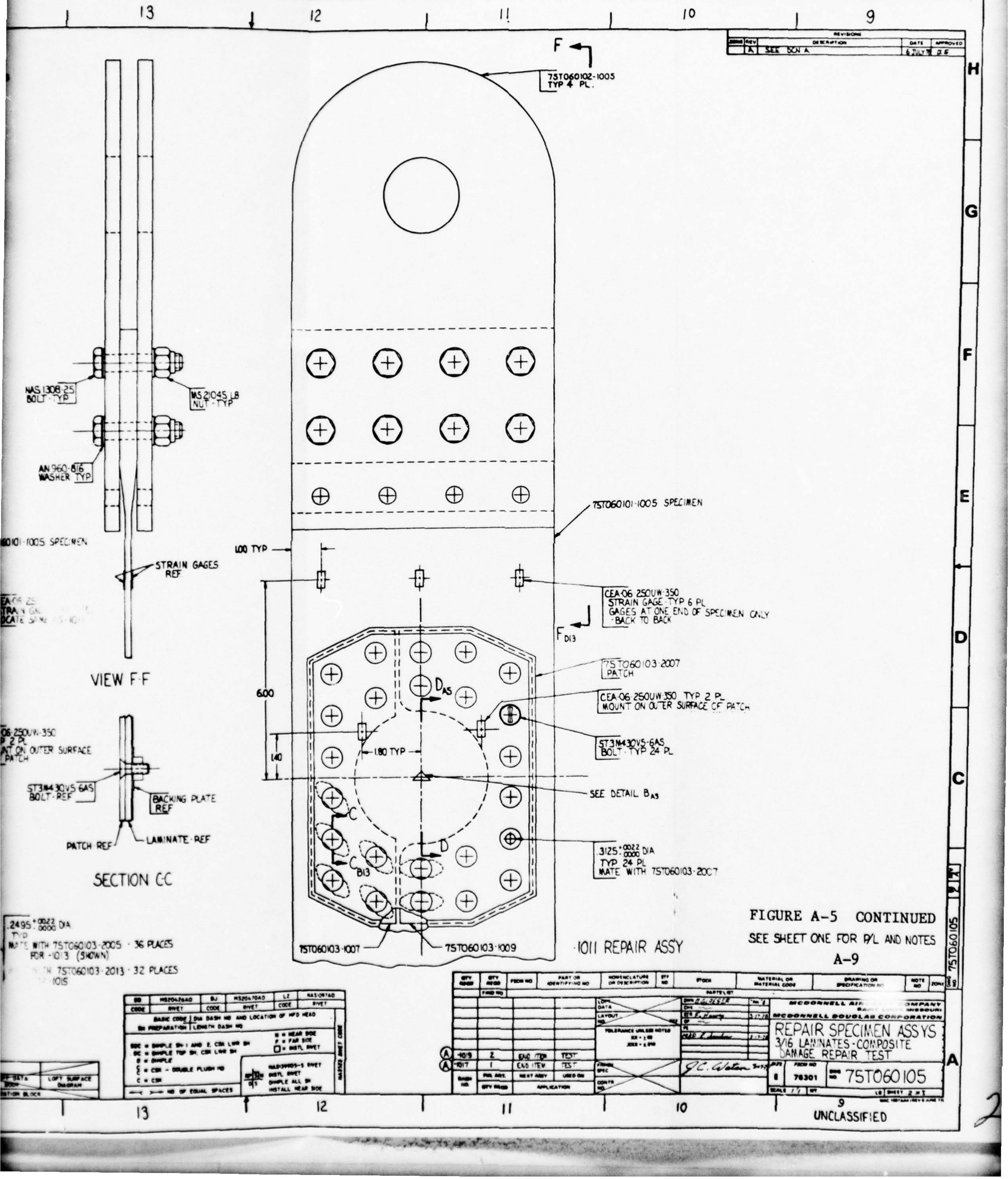
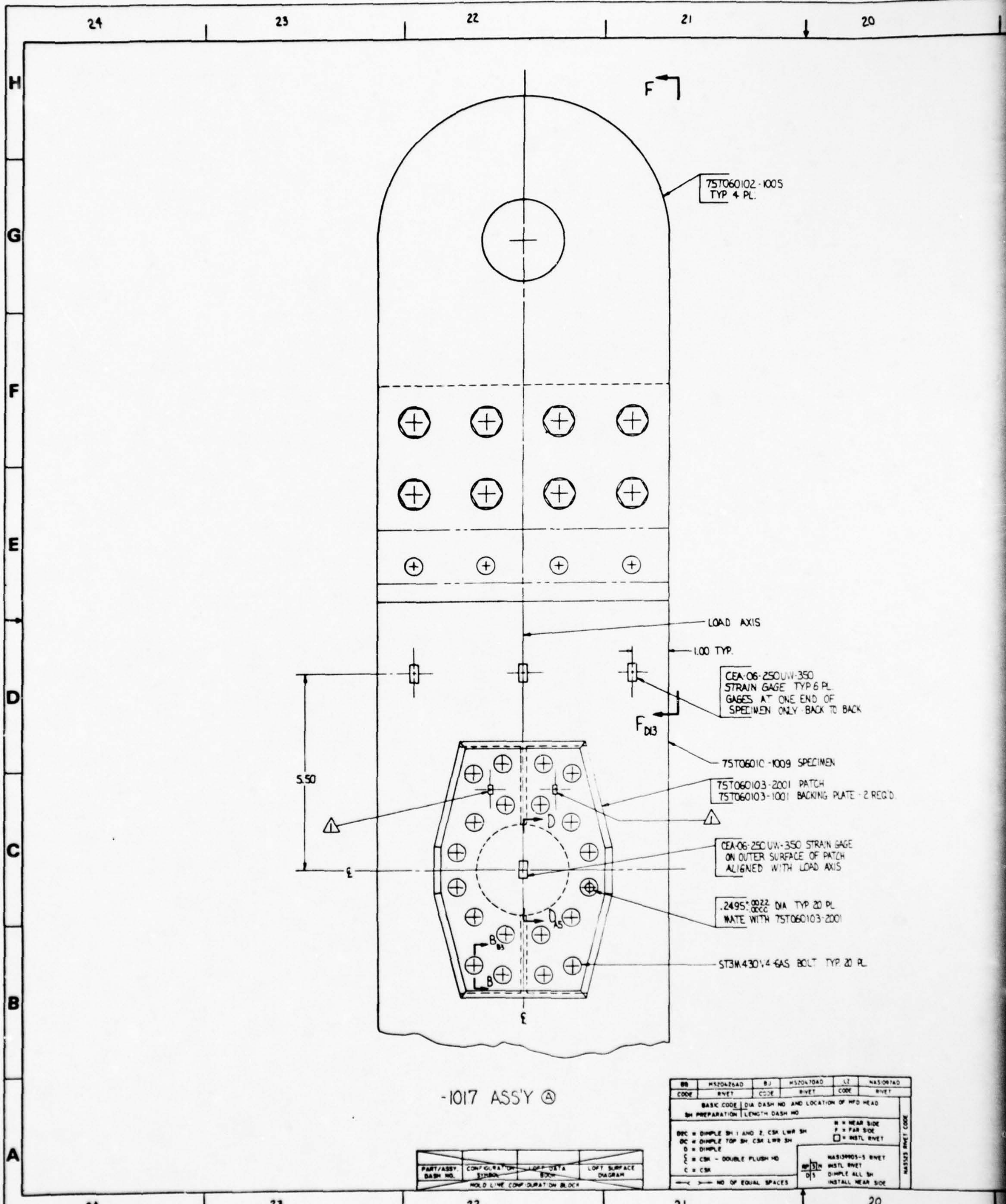


FIGURE A-5 CONTINUED

A-9

UNCLASSIFIED



F
75T060102-1005
TYP 4 PL.
LOAD AXIS
1.00 TYP.
CEA-06-250UW-350
STRAIN GAGE TYP 6 PL
GAGES AT ONE END OF
SPECIMEN ONLY - BACK TO BACK
F D13
75T06010-1009 SPECIMEN
5.50
75T060103-2001 PATCH
75T060103-1001 BACKING PLATE - 2 REQ'D
D
CEA-06-250UW-350 STRAIN GAGE
ON OUTER SURFACE OF PATCH
ALIGNED WITH LOAD AXIS
.2495 +.0022 -.0000 DIA TYP 20 PL
MATE WITH 75T060103-2001
D A5
B B3
B
ST3M430V4-6AS BOLT TYP 20 PL
-1017 ASS'Y Ⓐ
PART/ASSY. DASH NO.
CONFIGURATION SYMBOL
LOFT DATA BOOK
LOFT SURFACE DIAGRAM
MOLD LINE CONFIGURATION BLOCK
BB MS20426AD
CODE RIVET
BJ MS20470AD
CODE RIVET
LZ NAS1097AD
CODE RIVET
BASIC CODE DIA DASH NO AND LOCATION OF MFD HEAD
SH PREPARATION LENGTH DASH NO
DDC = DIMPLE SH 1 AND 2, CSK LWR SH
DC = DIMPLE TOP SH, CSK LWR SH
D = DIMPLE
C C = CSK - DOUBLE FLUSH HD
C = CSK
N = NEAR SIDE
F = FAR SIDE
= INSTL RIVET
NAS1399D5-5 RIVET
INSTL RIVET
DIMPLE ALL SH
INSTALL NEAR SIDE
NO OF EQUAL SPACES
NAS523 RIVET CODE

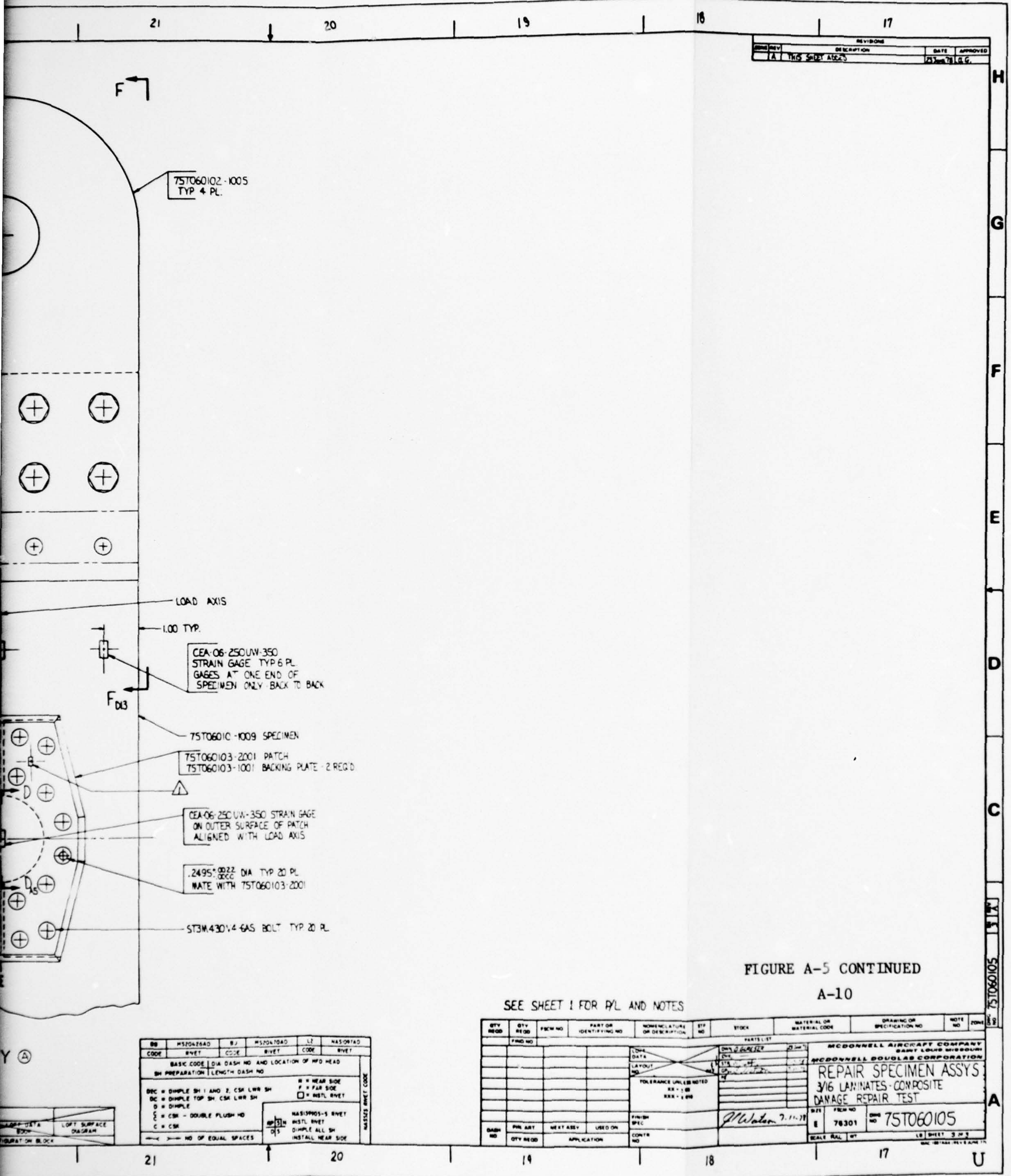
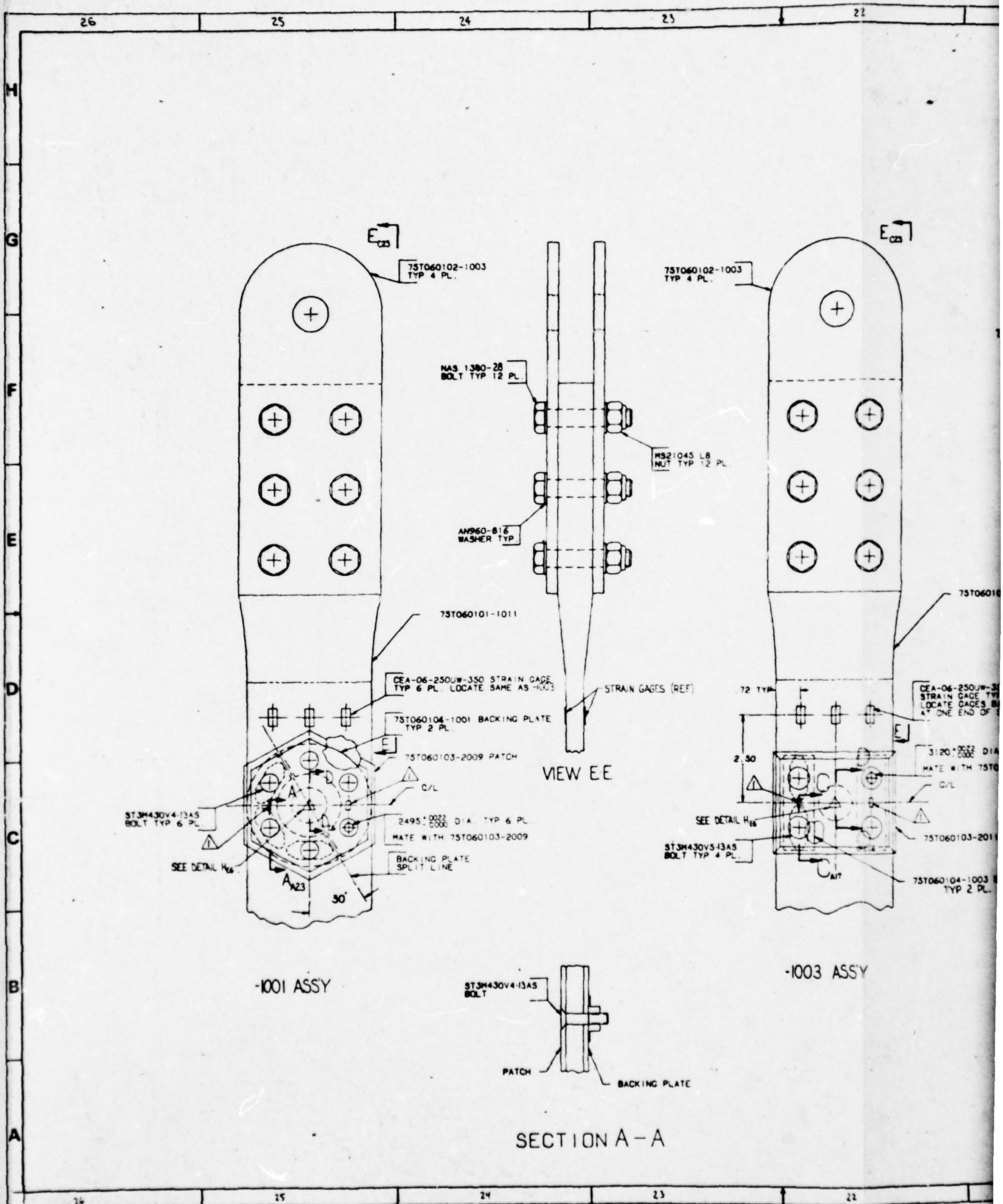
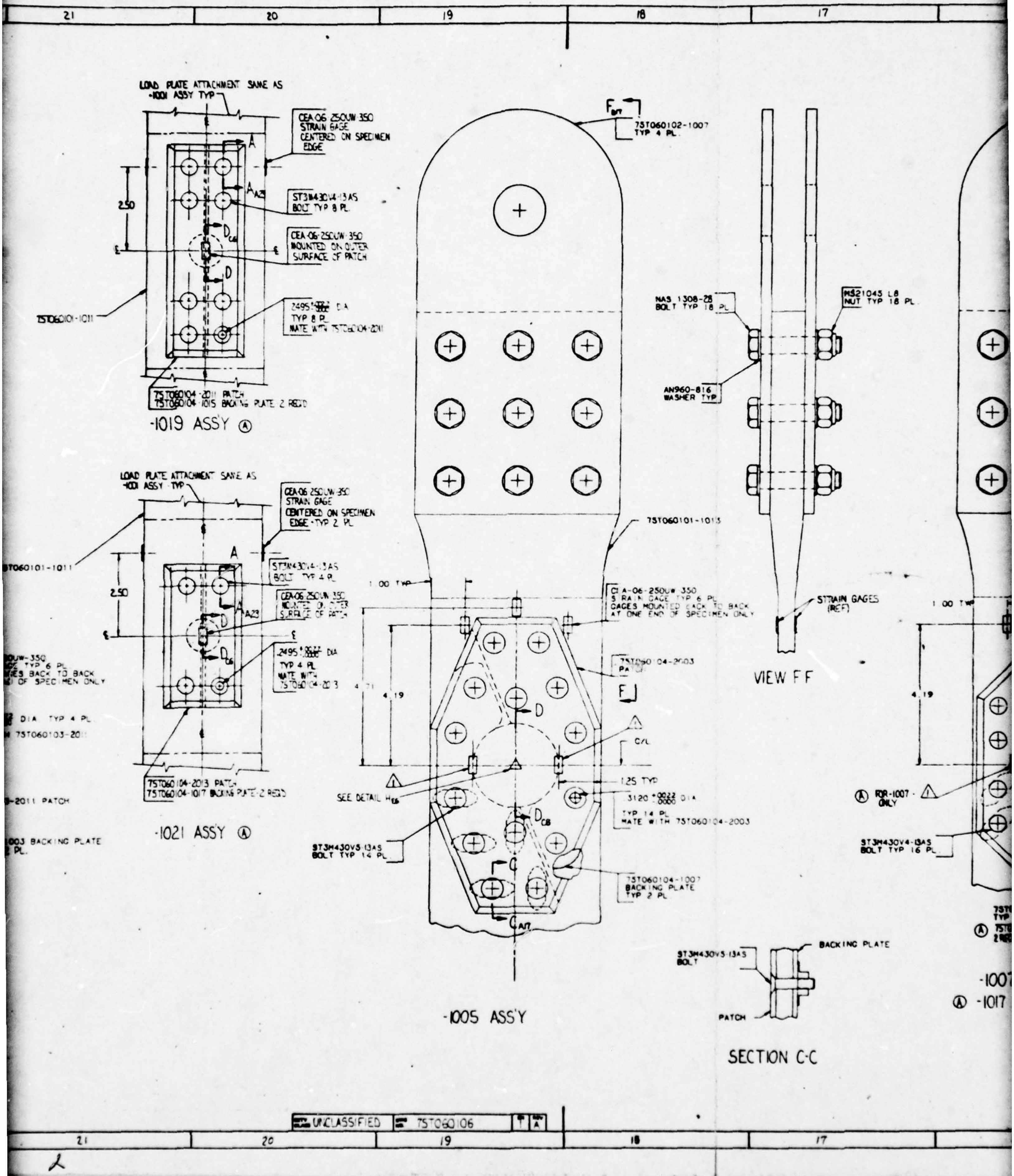


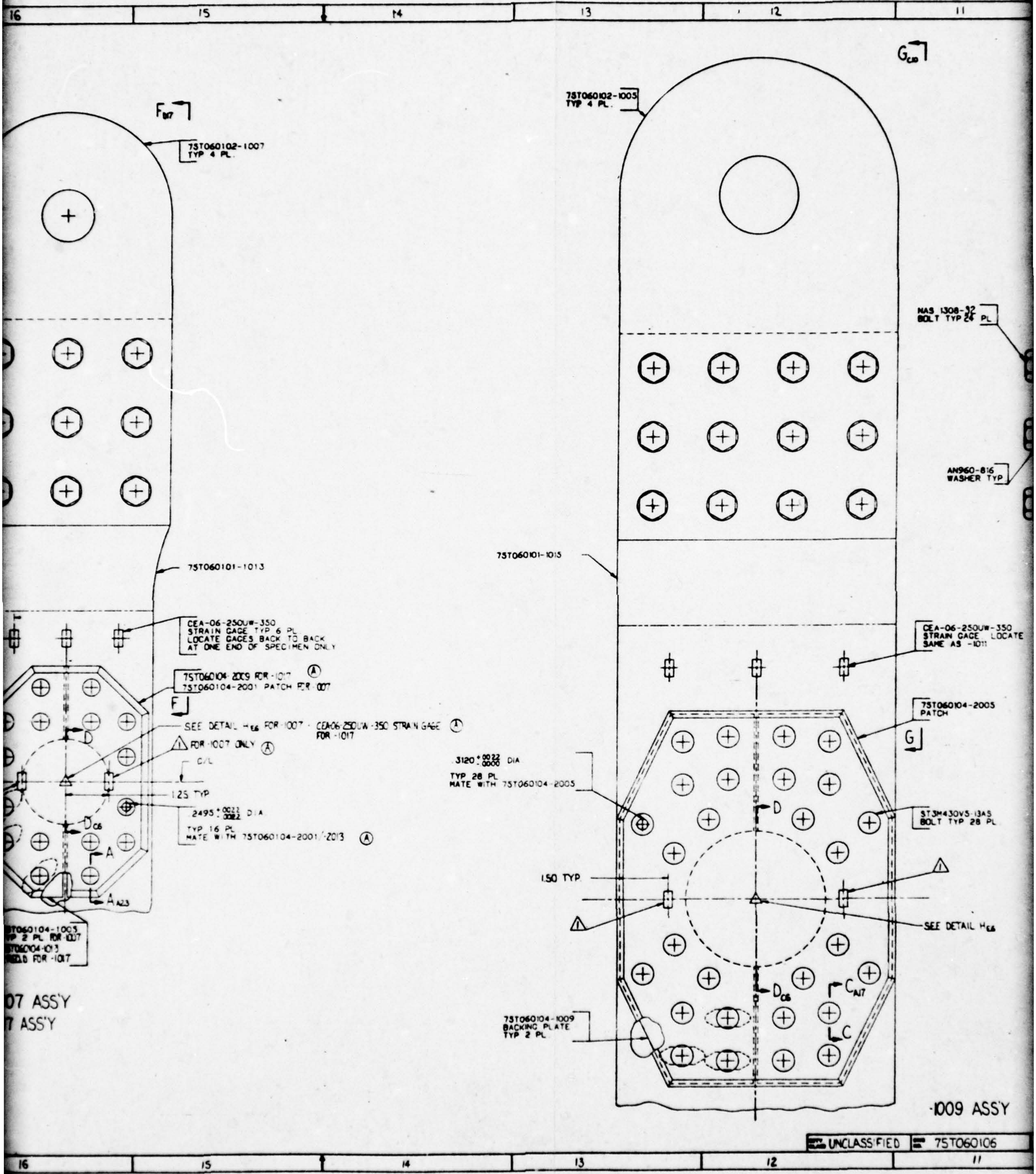
FIGURE A-5 CONTINUED



75T060102-1003
TYP 4 PL.
NAS 1380-28
BOLT TYP 12 PL.
MS21045 L8
NUT TYP 12 PL.
AN960-816
WASHER TYP.
75T060101-1011
CEA-06-250UW-350 STRAIN GAGE
TYP 6 PL. LOCATE SAME AS -1003
STRAIN GAGES (REF)
75T060104-1001 BACKING PLATE
TYP 2 PL.
75T060103-2009 PATCH
VIEW E-E
C/L
ST3M430V4-13AS
BOLT TYP 6 PL.
SEE DETAIL H
MATE WITH 75T060103-2009
BACKING PLATE
SPLIT LINE
30°
-1001 ASSY
.72 TYP
2.50
ST3M430V5-13AS
BOLT TYP 4 PL.
75T060104-1003
TYP 2 PL.
-1003 ASSY
ST3M430V4-13AS
BOLT
PATCH
BACKING PLATE
SECTION A-A

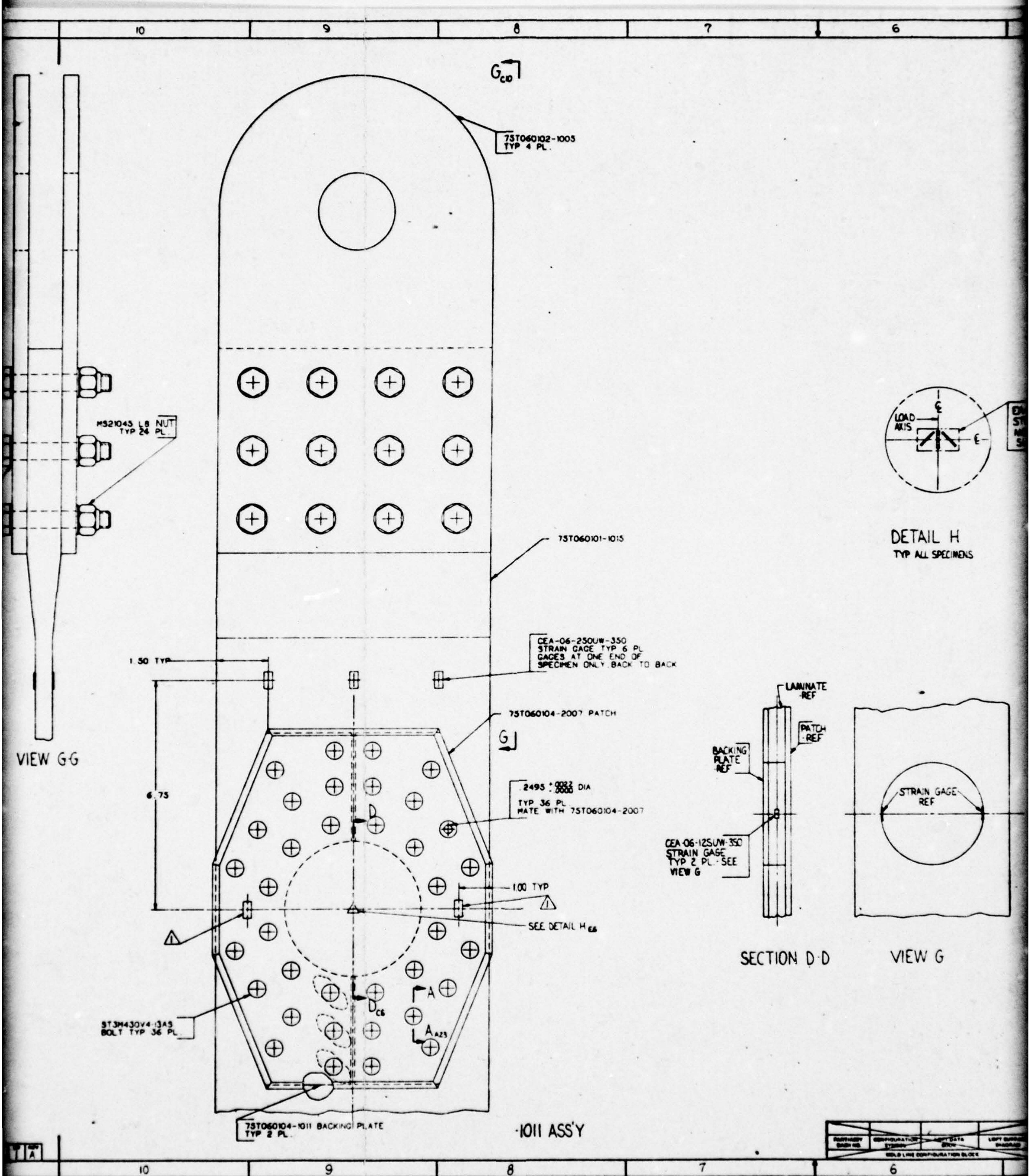


21
20
19
18
17
LOAD PLATE ATTACHMENT SAME AS -1001 ASSY TYP
CEA-06-250UW-350 STRAIN GAGE CENTERED ON SPECIMEN EDGE
ST3M430V4-13AS BOLT TYP 8 PL.
CEA-06-250UW-350 MOUNTED ON OUTER SURFACE OF PATCH
7ST060101-1011
TYP 8 PL MATE WITH 7ST060104-2011
7ST060104-2011 PATCH
7ST060104-1015 BACKING PLATE 2 REQD
-1019 ASSY Ⓐ
LOAD PLATE ATTACHMENT SAME AS -1001 ASSY TYP
CEA-06-250UW-350 STRAIN GAGE CENTERED ON SPECIMEN EDGE - TYP 2 PL.
ST3M430V4-13AS BOLT TYP 4 PL
CEA-06-250UW-350 MOUNTED ON OUTER SURFACE OF PATCH
TYP 4 PL MATE WITH 7ST060104-2013
7ST060104-2013 PATCH
7ST060104-1017 BACKING PLATE 2 REQD
-1021 ASSY Ⓐ
7ST060102-1007 TYP 4 PL.
7ST060101-1013
1.00 TYP
CEA-06-250UW-350 STRAIN GAGE TYP 6 PL GAGES MOUNTED BACK TO BACK AT ONE END OF SPECIMEN ONLY
7ST060104-2003 PATCH
4.19
1.25 TYP
SEE DETAIL H
.3120 +.0022 -.0000 DIA
TYP 14 PL MATE WITH 7ST060104-2003
ST3M430V5-13AS BOLT TYP 14 PL.
7ST060104-1007 BACKING PLATE TYP 2 PL.
-1005 ASS'Y
NAS 1308-28 BOLT TYP 18 PL
MS21045 L8 NUT TYP 18 PL.
AN960-816 WASHER TYP
STRAIN GAGES (REF)
VIEW F-F
ST3M430V5-13AS BOLT
BACKING PLATE
PATCH
SECTION C-C
Ⓐ FOR -1007 ONLY
ST3M430V4-13AS BOLT TYP 16 PL.
-1007
Ⓐ -1017
UNCLASSIFIED
7ST060106



UNCLASSIFIED | 7ST060106

3


10
9
8
7
6
G_{C10}
75T060102-1005
TYP 4 PL.
MS21045 L8 NUT
TYP 24 PL.
75T060101-1015
DETAIL H
TYP ALL SPECIMENS
LOAD AXIS
CEA-06-250UW-350
STRAIN GAGE TYP 6 PL
GAGES AT ONE END OF
SPECIMEN ONLY BACK TO BACK
1.50 TYP
75T060104-2007 PATCH
VIEW G-G
6.75
.2495 +.0022 -.0000 DIA
TYP 36 PL
MATE WITH 75T060104-2007
1.00 TYP
SEE DETAIL H_{E6}
D_{C6}
A_{A23}
ST3M430V4-13A3
BOLT TYP 36 PL
75T060104-1011 BACKING PLATE
TYP 2 PL.
-1011 ASS'Y
LAMINATE REF
PATCH REF
BACKING PLATE REF
CEA-06-125UW-350
STRAIN GAGE
TYP 2 PL. SEE
VIEW G
STRAIN GAGE REF
SECTION D-D
VIEW G
10
9
8
7
6

4

REVISIONS

REV	DESCRIPTION	DATE	APPROVED
A	SEE ECN A	[illegible]	[illegible]

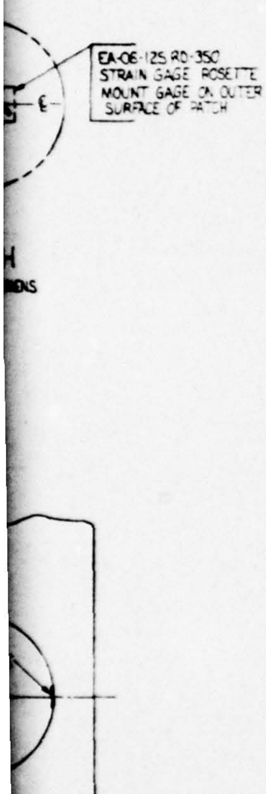
THIS PAGE IS BEST QUALITY PRACTICABLE
FROM COPY FURNISHED TO DDC

NOTES:

3. BOLTS MAY BE RE-USED ON MULTIPLE TEST ARTICLES.
2. ACCEPTABLE ALTERNATE ST3M571-8-32. USE WASHERS TO COMPENSATE FOR ANY EXCESS BOLT LENGTH.
1. CEA-06-125UW-350 STRAIN GAGE MOUNTED ON OUTER SURFACE OF PATCH. TYP 2 PLACES AS SHOWN.

FIGURE A-6 DETAILED DRAWING OF REPAIR ASSEMBLIES FOR 1/2 LAMINATES

A-11



			QTY REQD		QTY REQD		QTY REQD		QTY REQD		PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	NOTE NO
									2		75T060104-1017	BACKING PLATE	
										2	75T060104-1015	BACKING PLATE	
									1		75T060104-2013	PATCH	
										1	75T060104-2011	PATCH	
									X		-1021	REPAIR ASSY	
										X	-1019	REPAIR ASSY	
		1									75T060101-1019	SPECIMEN	
	1										75T060101-1017	SPECIMEN	
2											75T060104-1013	BACKING PLATE	
1											75T060104-2009	PATCH	
X											-1017	REPAIR ASSY	
	X										-1015	REPAIR ASSY	
		X									-1013	REPAIR ASSY	
-	-	-	1	1	1	1	1	1	-	-	EA-06-125RD-350	STRAIN GAGE	
2	2	2	4	4	4	4	4	4	-	-	CEA-06-125UW-350	STRAIN GAGE	
7	7	7	6	6	6	6	6	6	3	3	CEA-06-250UW-350	STRAIN GAGE	
AR	AR	AR	AR	AR	AR	AR	AR	AR	AR	AR	AN960-816	WASHER	
18	24	24	24	24	18	18	12	12	12	12	MS21045 L8	NUT	
	24	24	24	24							NAS1308-32	BOLT	2,3
8					18	18	12	12	12	12	NAS1308-28	BOLT	2,3
				28		14	4				ST3M430V5-13AS	BOLT	
16	16	16	36		16			6	4	8	ST3M430V4-13AS	BOLT	
							1				75T060103-2011	PATCH	
								1			75T060103-2009		
			1								75T060104-2007		
				1							75T060104-2005		
						1					75T060104-2003		
	1	1			1						75T060104-2001	PATCH	
			2								75T060104-1011	BACKING PLATE	
				2							75T060104-1009		
						2					75T060104-1007		
	2	2			2						75T060104-1005		
							2				75T060104-1003		
								2			75T060104-1001	BACKING PLATE	
4					4	4					75T060102-1007	LOAD PLATE	
	4	4	4	4							75T060102-1005	LOAD PLATE	
							4	4	4	4	75T060102-1003	LOAD PLATE	
			1	1							75T060101-1015	SPECIMEN	
1					1	1					75T060101-1013	SPECIMEN	
							1	1	1	1	75T060101-1011	SPECIMEN	
			X								-1011	REPAIR ASSY	
				X							-1009		
					X						-1007		
						X					-1005		
							X				-1003		
								X			-1001	REPAIR ASSY	

PARTS LIST

A CONT'D ON SH 2

DASH NO	QTY REQD	NEXT ASSY	USED ON
-1013	1	END ITEM	TEST
-1011	3	END ITEM	TEST
-1009	1		
-1007	1		
-1005	1		
-1003	1		
-1001	1	END ITEM	TEST

APPLICATION

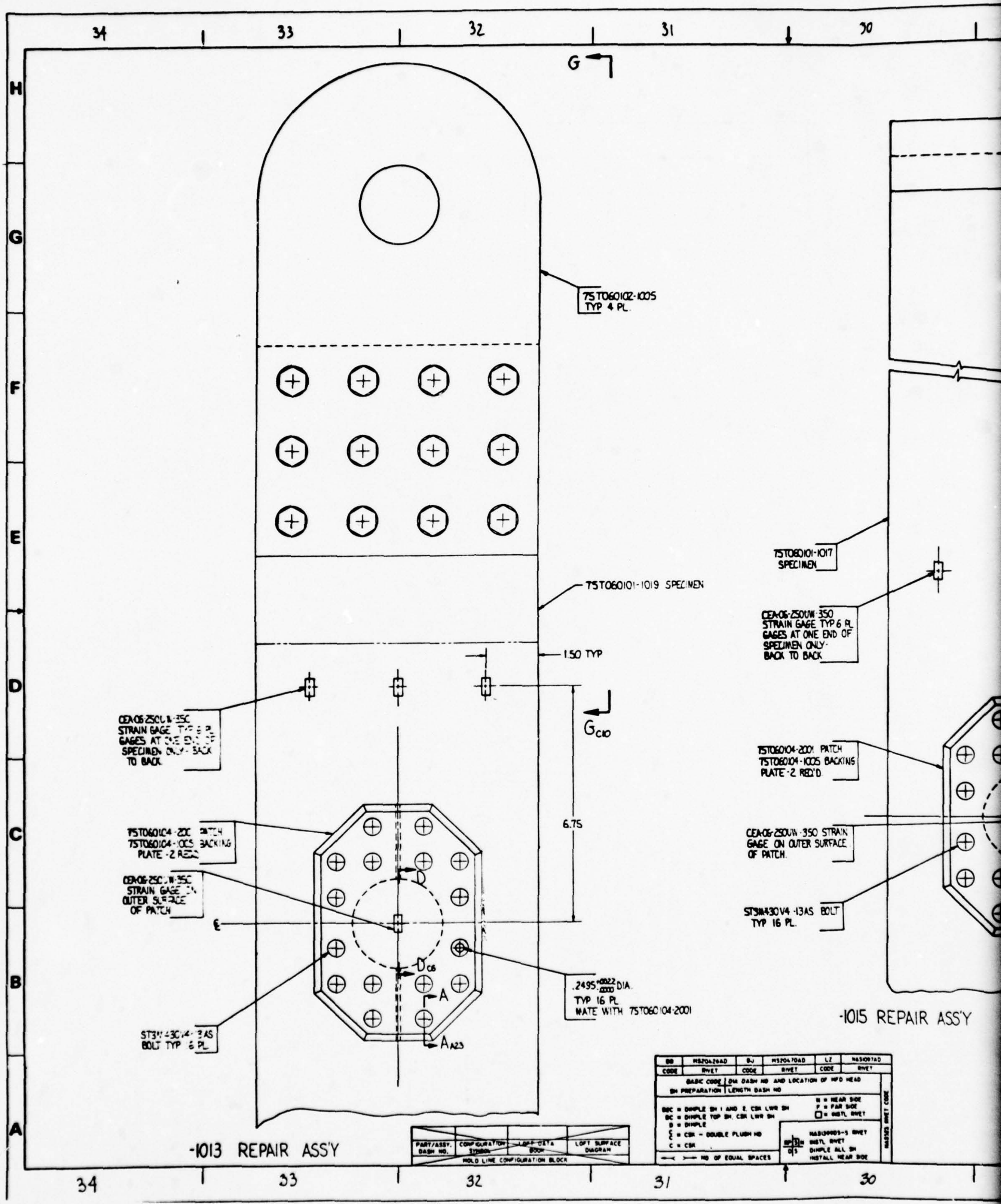
MCDONNELL AIRCRAFT COMPANY
MCDONNELL DOUGLAS CORPORATION

REPAIR SPECIMEN ASSYS
1/2 LAMINATES - COMPOSITE
DAMAGE REPAIR TESTS

SIZE J | CODE 76301 | 75T060106

UNCLASSIFIED | 75T060106 | REV A

75T060106



34
33
32
31
30
H
G
F
E
D
C
B
A
G
7ST060102-1005
TYP 4 PL.
7ST060101-1019 SPECIMEN
1.50 TYP
6.75
G_{C10}
CEA-06-250UW-350
STRAIN GAGE TYP 6 PL
GAGES AT ONE END OF
SPECIMEN ONLY - BACK
TO BACK
7ST060104-2001 PATCH
7ST060104-1005 BACKING
PLATE - 2 REQ'D
CEA-06-250UW-350
STRAIN GAGE ON
OUTER SURFACE
OF PATCH
E
D
D_{C6}
A
A_{A23}
ST3M430V4-13AS
BOLT TYP 16 PL.
.2495 $^{+.0022}_{-.0000}$ DIA.
TYP 16 PL
MATE WITH 7ST060104-2001
-1013 REPAIR ASS'Y
7ST060101-1017
SPECIMEN
CEA-06-250UW-350
STRAIN GAGE TYP 6 PL
GAGES AT ONE END OF
SPECIMEN ONLY -
BACK TO BACK
7ST060104-2001 PATCH
7ST060104-1005 BACKING
PLATE - 2 REQ'D
CEA-06-250UW-350 STRAIN
GAGE ON OUTER SURFACE
OF PATCH.
ST3M430V4-13AS BOLT
TYP 16 PL.
-1015 REPAIR ASS'Y
PART/ASSY. DASH NO.
CONFIGURATION SYMBOL
LOFT DATA BOOK
LOFT SURFACE DIAGRAM
MOLD LINE CONFIGURATION BLOCK
BB MS20426AD
CODE RIVET
BJ MS20470AD
CODE RIVET
LZ NAS1097AD
CODE RIVET
BASIC CODE DIA DASH NO AND LOCATION OF MFD HEAD
SH PREPARATION LENGTH DASH NO
DDC = DIMPLE SH 1 AND 2, CSK LWR SH
DC = DIMPLE TOP SH, CSK LWR SH
D = DIMPLE
C = CSK - DOUBLE FLUSH NO
C = CSK
N = NEAR SIDE
F = FAR SIDE
= INSTL RIVET
NAS1399D5-5 RIVET
INSTL RIVET
DIMPLE ALL SH
INSTALL NEAR SIDE
NO OF EQUAL SPACES
NAS523 RIVET CODE

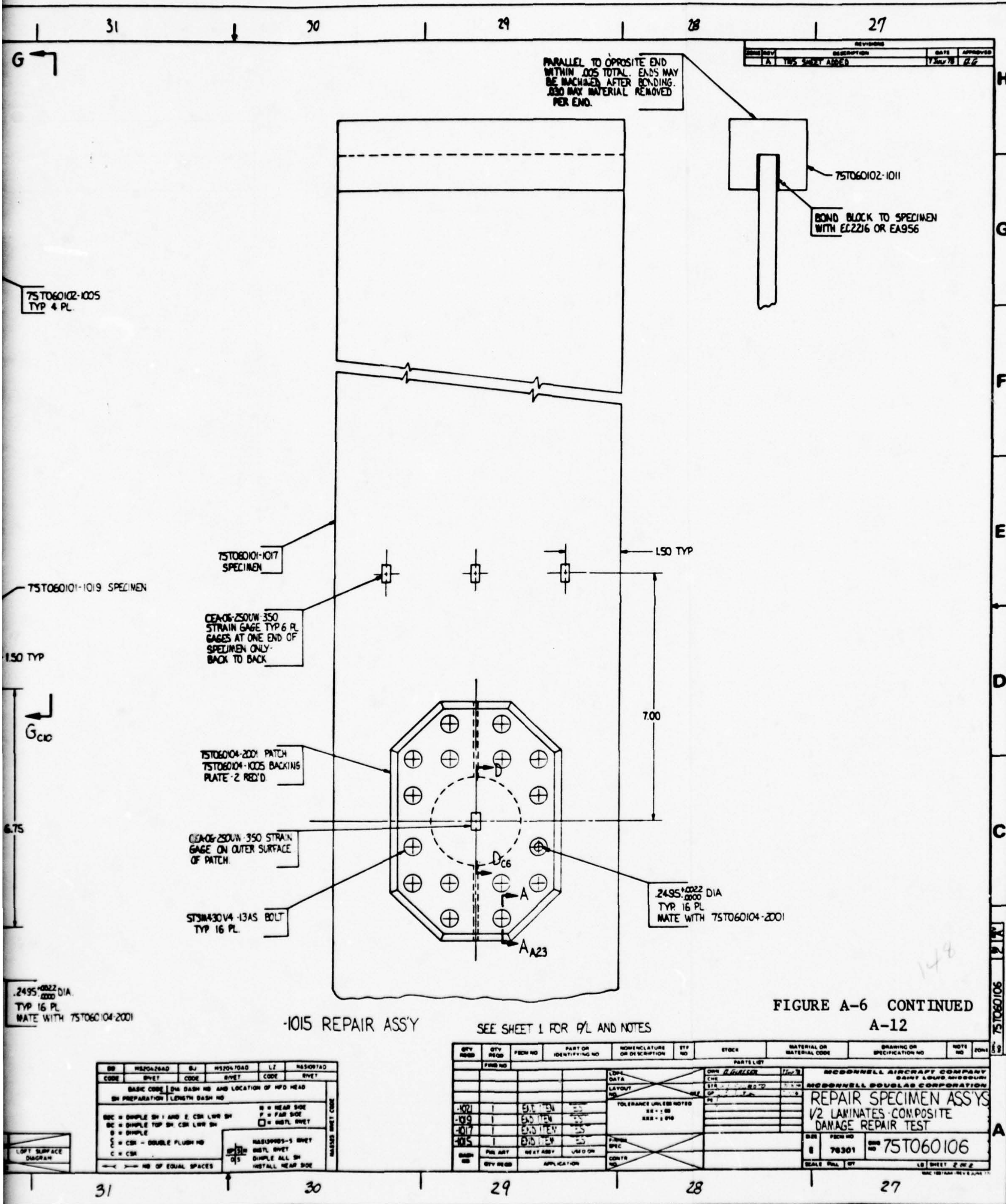
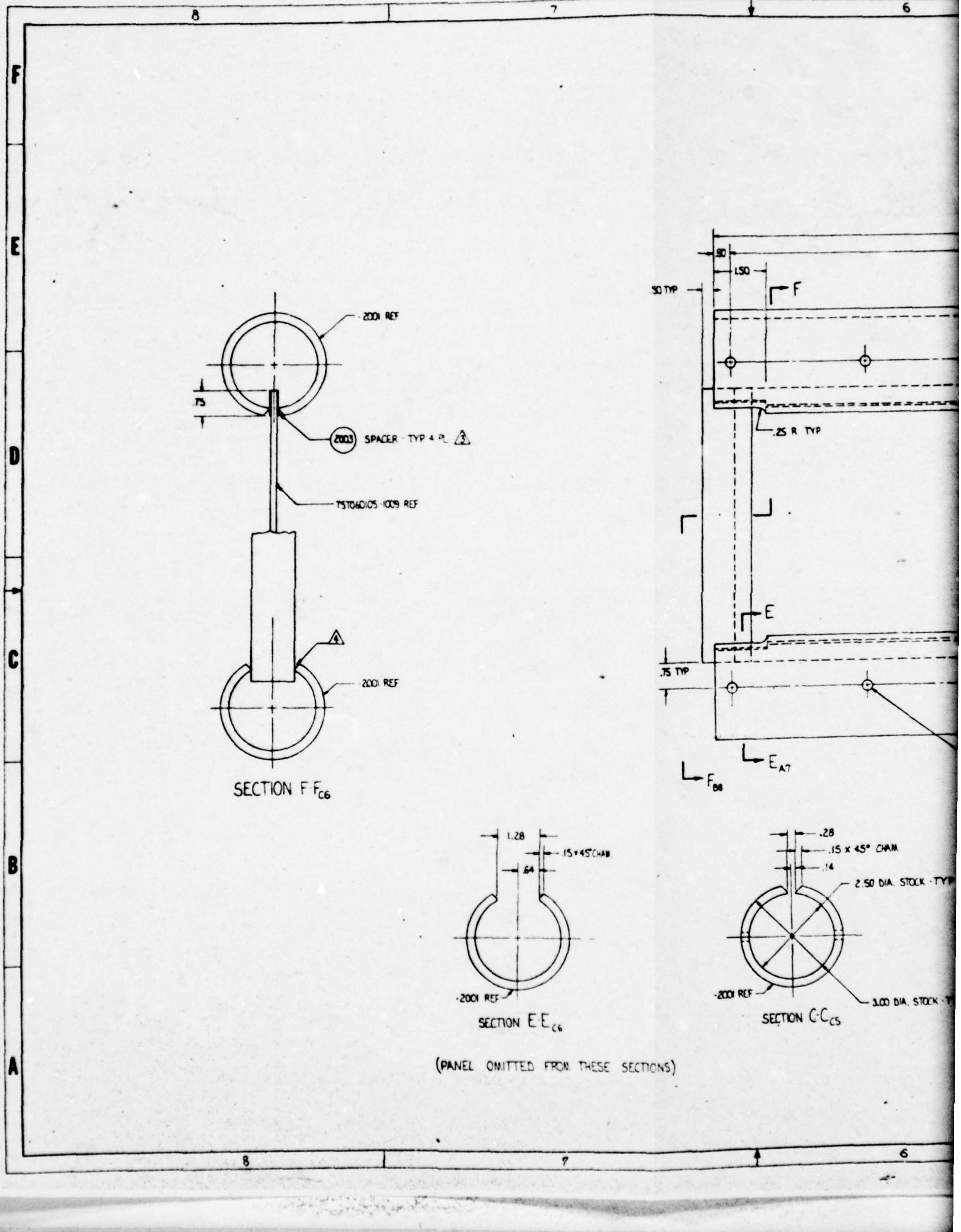
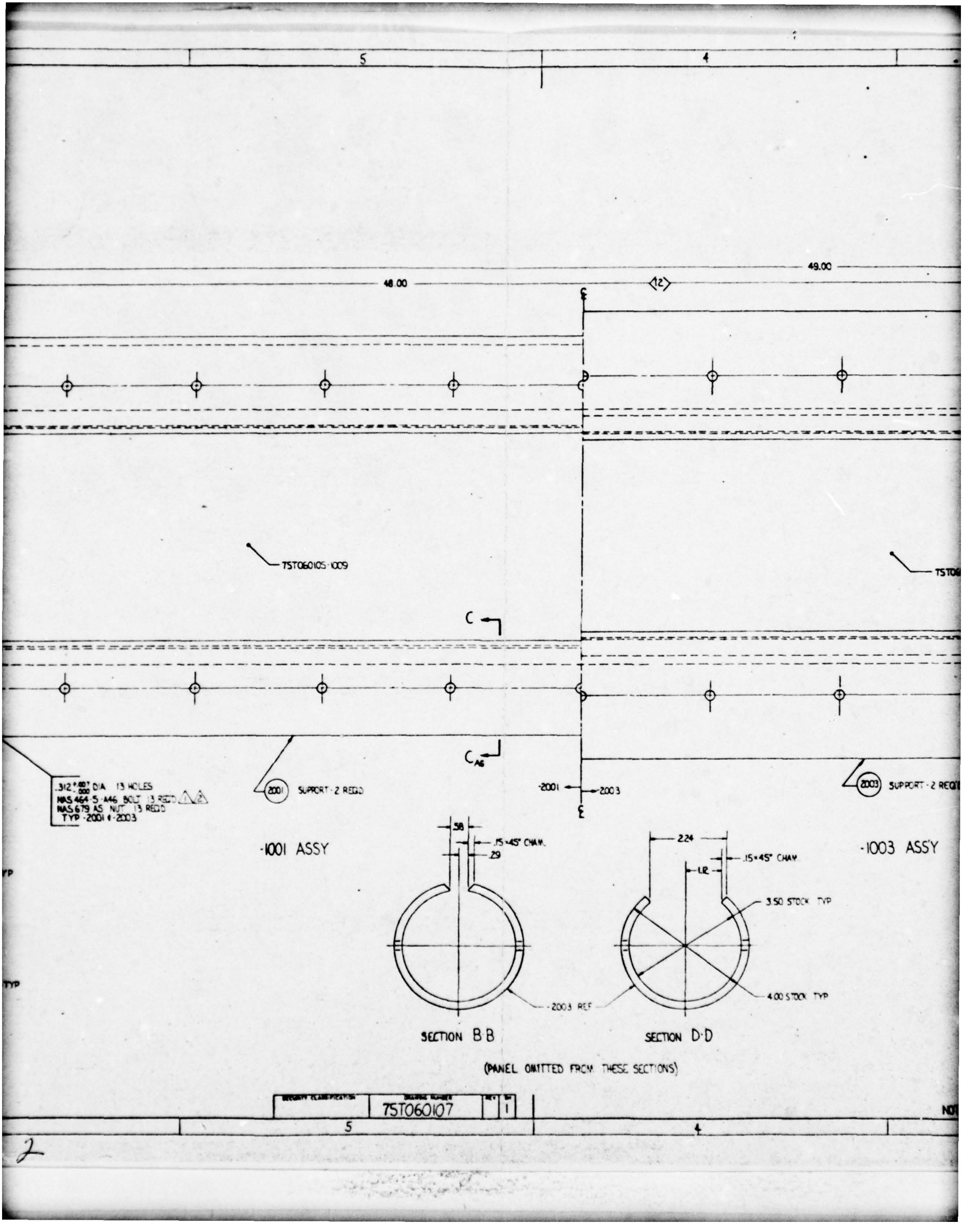


FIGURE A-6 CONTINUED



8
7
6
F
E
D
C
B
A
2001 REF
.75
2003 SPACER - TYP 4 PL 3
TST060105-1009 REF
4
2001 REF
SECTION F-F C6
.90
1.50
.50 TYP
F
.25 R TYP
E
.75 TYP
E A7
F B8
1.28
.15 × 45° CHAM.
.64
2001 REF
SECTION E-E C6
.28
.15 × 45° CHAM.
.14
2.50 DIA. STOCK - TYP
2001 REF
3.00 DIA. STOCK - TY
SECTION C-C C5
(PANEL OMITTED FROM THESE SECTIONS)


5
4
48.00
49.00
12
7ST060105-1009
7ST06
C
C A6
.312 +.007 -.000 DIA 13 HOLES
NAS 464-5-A46 BOLT 13 REQD 1 2
NAS 679 A5 NUT 13 REQD
TYP -2001 & -2003
2001 SUPPORT - 2 REQD
-2001
-2003
2003 SUPPORT - 2 REQD
-1001 ASSY
-1003 ASSY
.58
.15 × 45° CHAM.
.29
2.24
1.12
3.50 STOCK TYP
4.00 STOCK TYP
-2003 REF
SECTION B-B
SECTION D-D
(PANEL OMITTED FROM THESE SECTIONS)
TYP
7ST060107
1
2

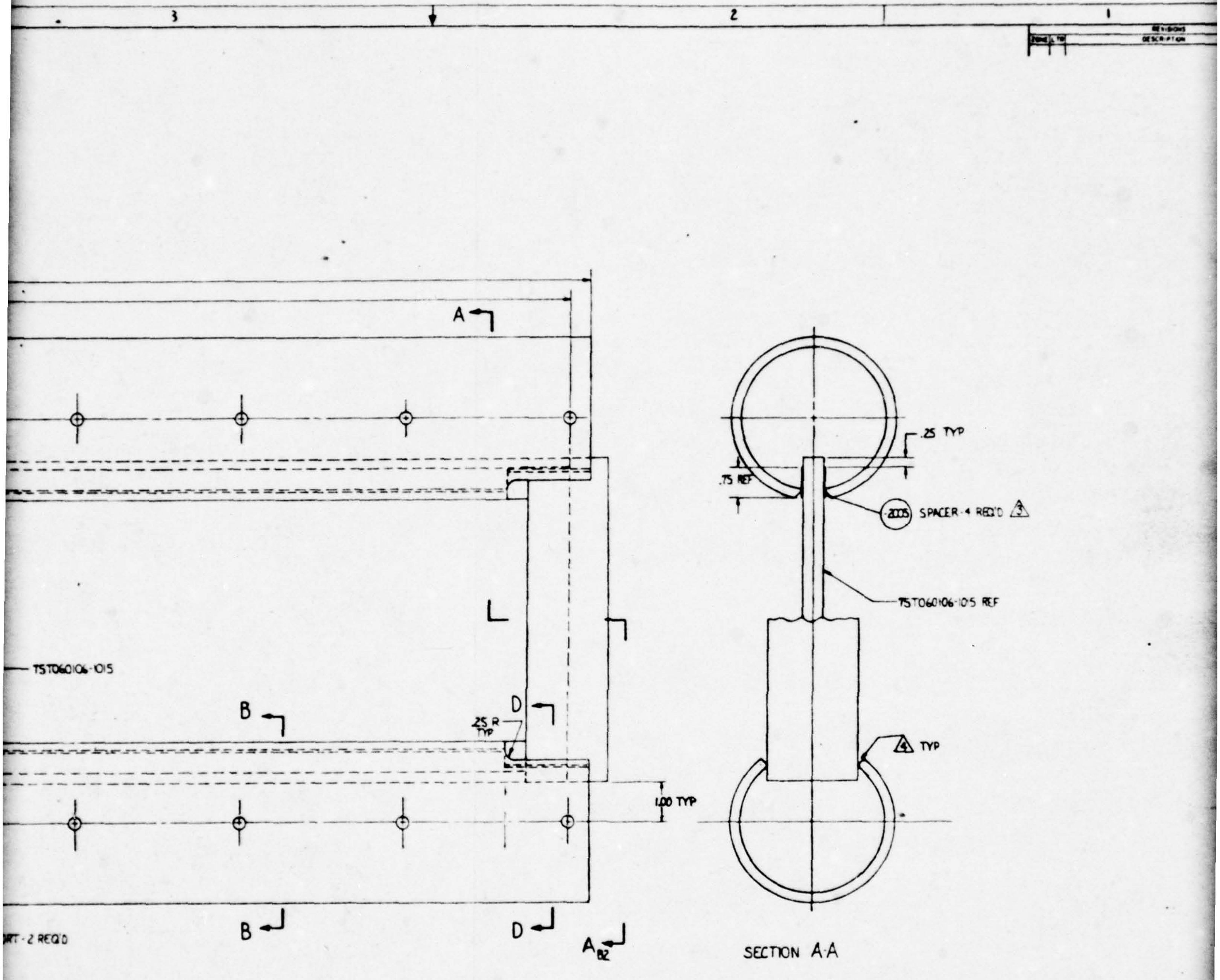


FIGURE A-7 DETAILED DRAWING OF EDGE SU
FOR COMPRESSION SPECIME

QTY REQD	QTY REQD	CODE IDENT NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE	DRAWING SPECIFICAT
1			75T060106-1015	REPAIR ASSY			
	1		75T060105-1009	REPAIR ASSY			
4	4		-2005	SPACER	.032 × .75 × 47.00	2024-T3	QQA 250/4
26	26		NAS679 A5	NUT			
26	26		NAS464 S A46	BOLT			
2			-2003	SUPPORT	4.000 × .25 WALL × 49.00	BLACK PIPE	
	2		-2001	SUPPORT	5.000 × .25 WALL × 49.00	BLACK PIPE	
-1003	-1001						

PARTS LIST

-1003	1	ENG ITEM	TEST
-1001	1	ENG ITEM	TEST
DASH NO	FIN ART	NEXT ASSY	USED ON

TOLERANCE UNLESS NOTED
.XX = ±.03
.XXX = ±.010

MCDONNELL AI...
MCDONNELL DOUGL...

SUPPORT A...
COMPOSITE DAMAG...
COMPRES...

SIZE J CODE IDENT NO 76301 75T...

11 TEST LAB (DEPT 253) MAY HAVE MATERIAL IN STOCK (REF GLENN HARRISON × 25652)
10 INSTALL FASTENERS PER PS 19000
9 SURFACE ROUGHNESS 125 RHR OR BETTER UNLESS OTHERWISE NOTED
8 BREAK ALL SHARP EDGES PER PS 2304
7 MACHINE FINISH PER PS 23041
6 LAST VIEW LETTER USED IS F
5 MARKING PER PS 15001
4 PLACE TEFLON TAPE BETWEEN SUPPORT AND END PLATE
3 PLACE TEFLON TAPE BETWEEN PANEL AND SPACER
2 TIGHTEN TO STABILIZE PANEL. DO NOT TIGHTEN EXCESSIVELY-OVER CLAMPING MAY DAMAGE PANEL
NOTES: 1 INSTALLATION FASTENER

3 2 1

3

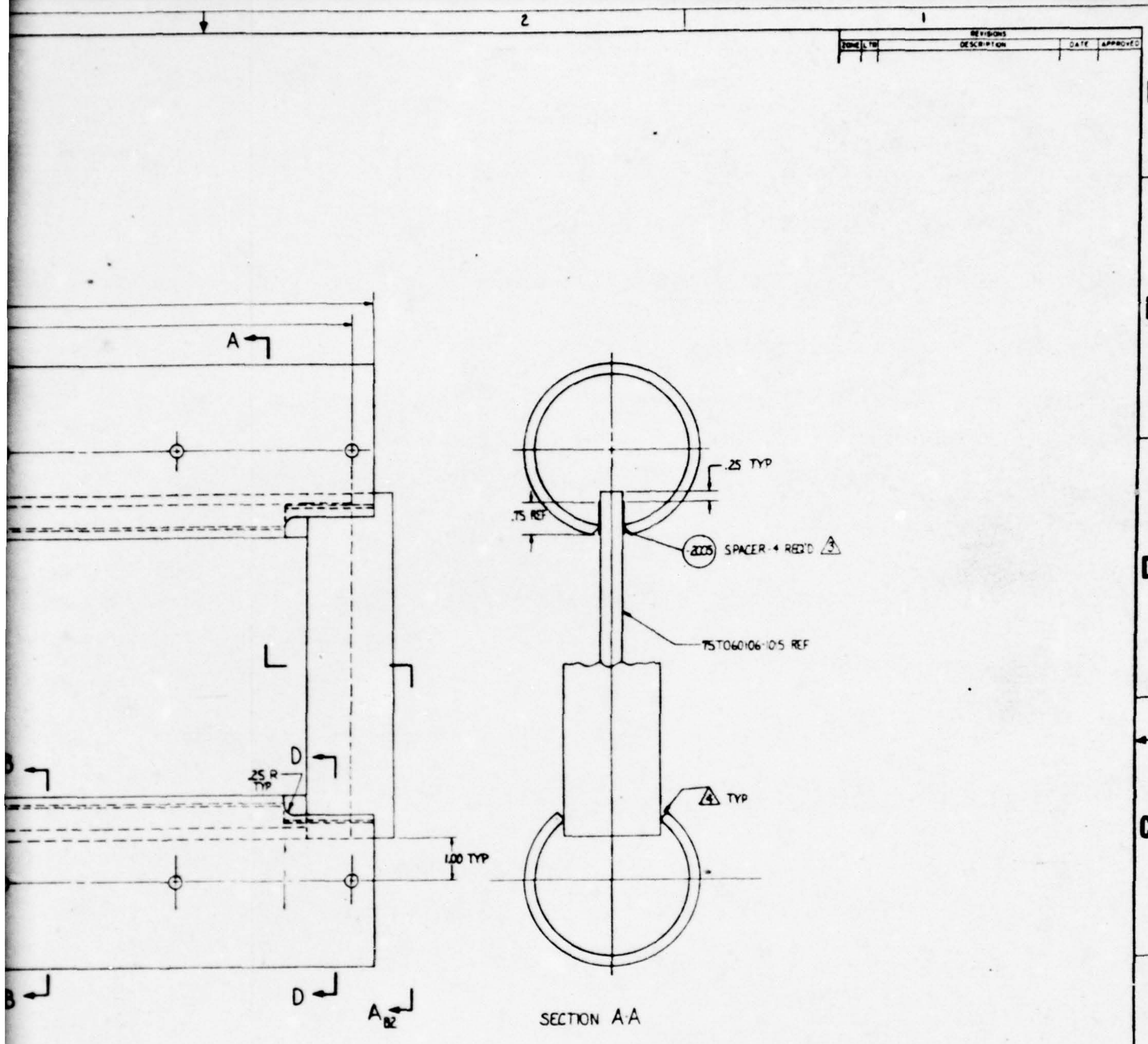


FIGURE A-7 DETAILED DRAWING OF EDGE SUPPORTS FOR COMPRESSION SPECIMENS

QTY REQD -1003	QTY REQD -1001	CODE IDENT NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	STOCK	MATERIAL OR MATERIAL CODE	DRAWING OR SPECIFICATION NO	NOTE NO	ZONE
1			7ST060106-1015	REPAIR ASSY					
	1		7ST060105-1009	REPAIR ASSY					
4	4		-2005	SPACER	.032 × .75 × 47.00	2024-T3	QQA 250/4 COND T3		
26	26		NAS679 A5	NUT					
26	26		NAS464 S A46	BOLT					
2			-2003	SUPPORT	4.00 OD × .25 WALL × 49.00	BLACK PIPE		11	
	2		-2001	SUPPORT	3.00 OD × .25 WALL × 49.00	BLACK PIPE		11	

PARTS LIST

DASH NO	QTY REQD	NEXT ASSY	USED ON
-1003	1	END ITEM	TEST
-1001	1	END ITEM	TEST

TOLERANCE UNLESS NOTED
.XX = ±.03
.XXX = ±.010

MCDONNELL AIRCRAFT COMPANY
SAINT LOUIS MISSOURI
MCDONNELL DOUGLAS CORPORATION

SUPPORT ASSEMBLY
COMPOSITE DAMAGE REPAIR
COMPRES

SIZE J | CODE IDENT NO 76301 | 7ST060107
SCALE FULL | SHEET 1 OF 1

...AY HAVE MATERIAL IN STOCK (REF GLENN HARRISON +25652)
... PS 19000
... RNR OR BETTER UNLESS OTHERWISE NOTED
... PER PS23C4
...S 23041
...15 F

...TWEEN SUPPORT AND END PLATE
...TWEEN PANEL AND SPACER
...PANEL. DO NOT TIGHTEN EXCESSIVELY-OVER CLAMPING MAY DAMAGE PANEL
...ER

4

APPENDIX B

STRAIN GAGE DATA

APPENDIX B

STRAIN GAGE DATA

Strain Gage Data for all tests conducted are presented herein. Refer to Figures 3-2, 3-3 and 3-4 for the locations of the strain gages on the specimens.

FIGURE NO	TEST SPECIMEN NO	DESCRIPTION OF SPECIMEN
B-1	75T060105-1001	3/16 Laminate
B-2	75T060105-1003	1.0 Diameter
B-3	75T060105-1019 #1	Damage Hole
B-4	75T060105-1019 #2	
B-5	75T060105-1005 #1	3/16 Laminate
B-6	75T060105-1007	2.50 Diameter
B-7	75T060105-1005 #2	Damage Hole
B-8	75T060105-1005 #3	
B-9	75T060105-1011	3/16 Laminate
B-10	75T060105-1013	4.0 Diameter
B-11	75T060105-1015 #1	Damage Hole
B-12	75T060105-1015 #2	
B-13	75T060105-1017	3/16 Laminate Off-Axis
B-14	75T060105-1009	3/16 Laminate Compression
B-15	75T060106-1019	1/2 Laminate
B-16	75T060106-1021 #1	1.0 Diameter
B-17	75T060106-1021 #2	Damage Hole
B-18	75T060106-1021 #3	
B-19	75T060106-1005	1/2 Laminate
B-20	75T060106-1007 #1	2.50 Diameter
B-21	75T060106-1007 #2	Damage Hole
B-22	75T060106-1017	

FIGURE NO	TEST SPECIMEN NO	DESCRIPTION OF SPECIMEN
B-23	75T060106-1009	1/2 Laminate
B-24	75T060106-1011 #1	4.0 Diameters
B-25	75T060106-1011 #2	Damage Hole
B-26	75T060106-1011 #3	
B-27	75T060106-1013	1/2 Laminate Off-Axis
B-26	75T060106-1015	1/2 Laminate Compression

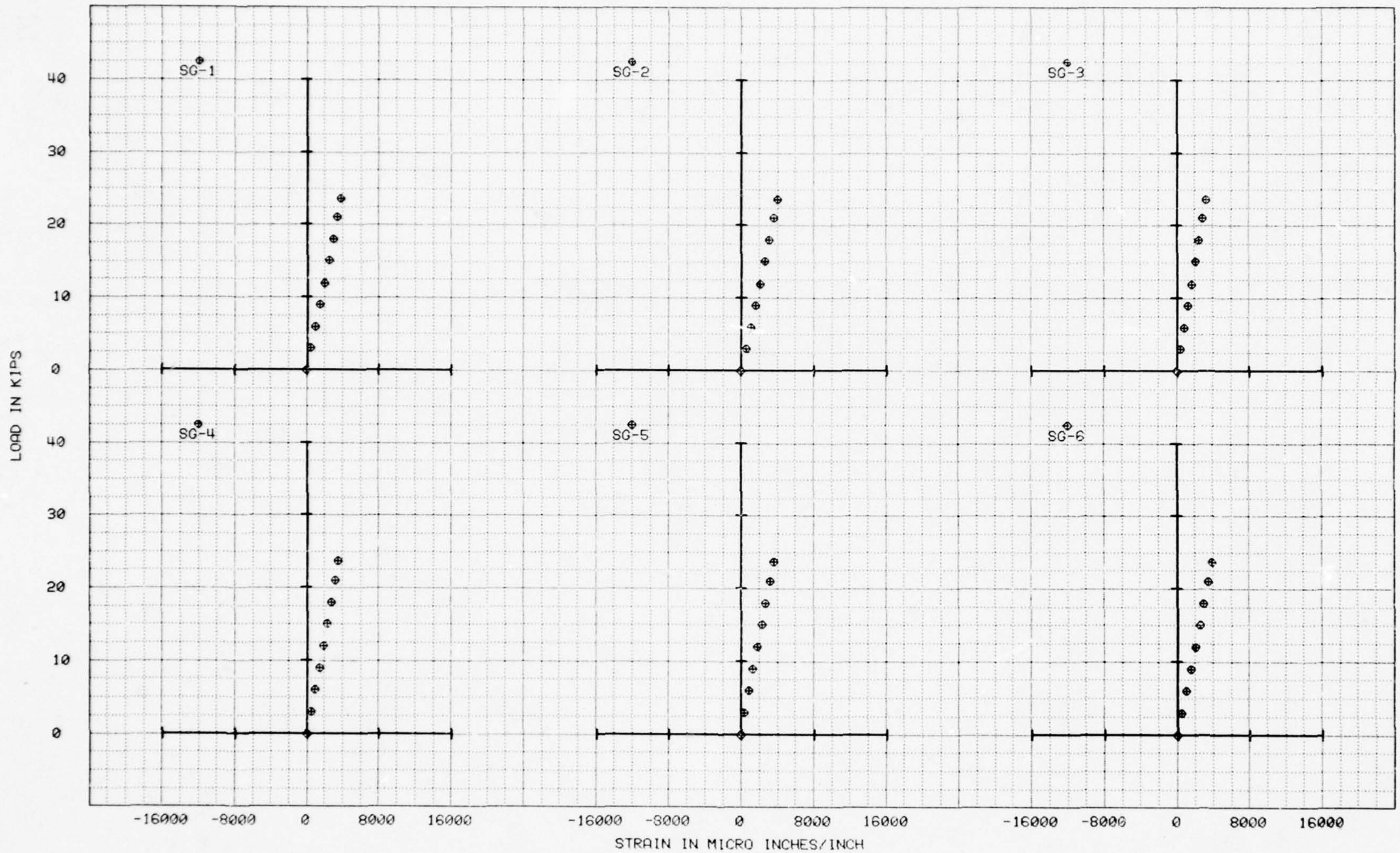


FIGURE B-1 Strain Gage Data 75T060105-1001
3/16 Laminate 1.0 Dia Damage Hole

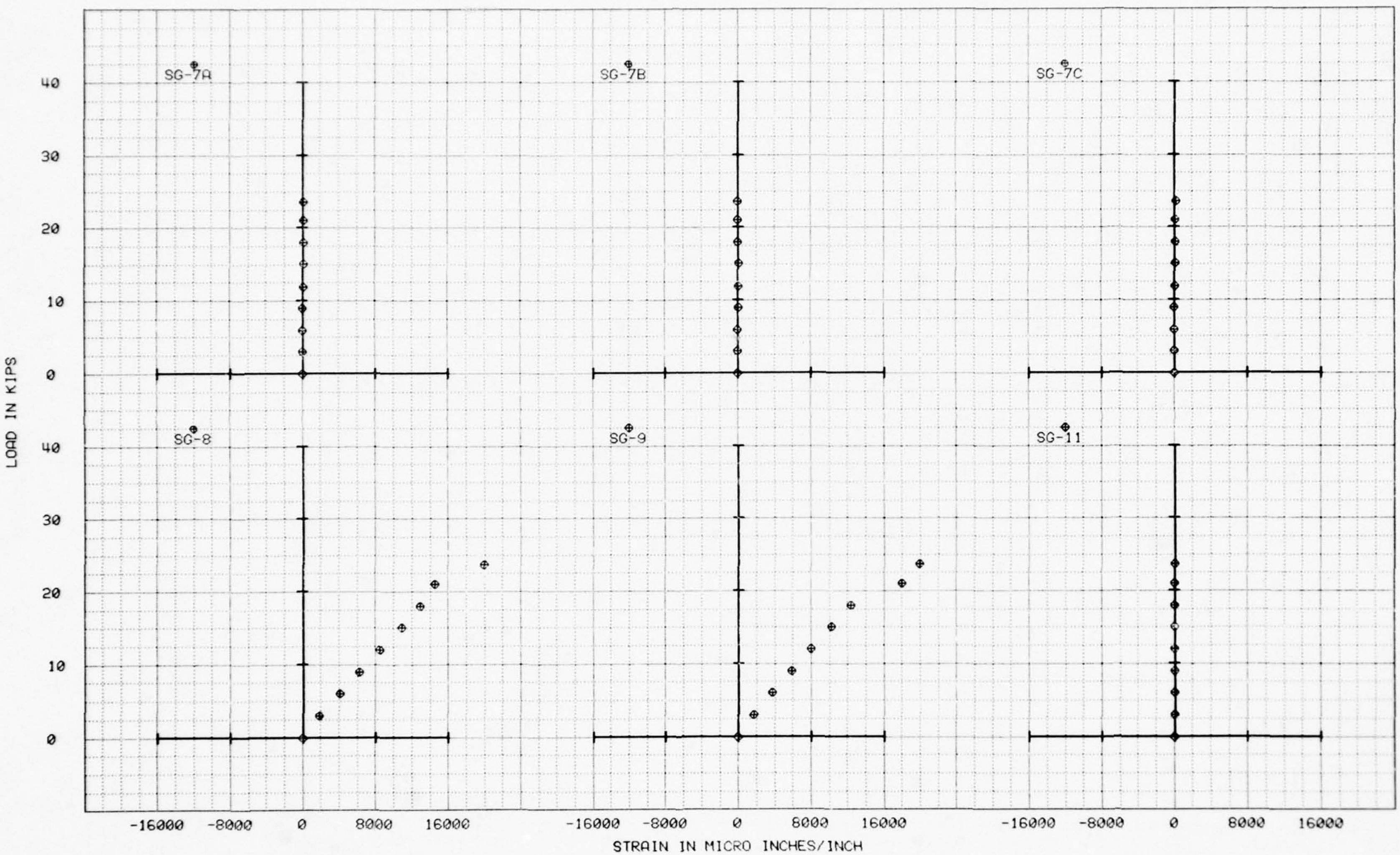


FIGURE B-1 Continued Strain Gage Data 75T060105-1001
3/16 Laminate 1.0 Dia Damage Hole

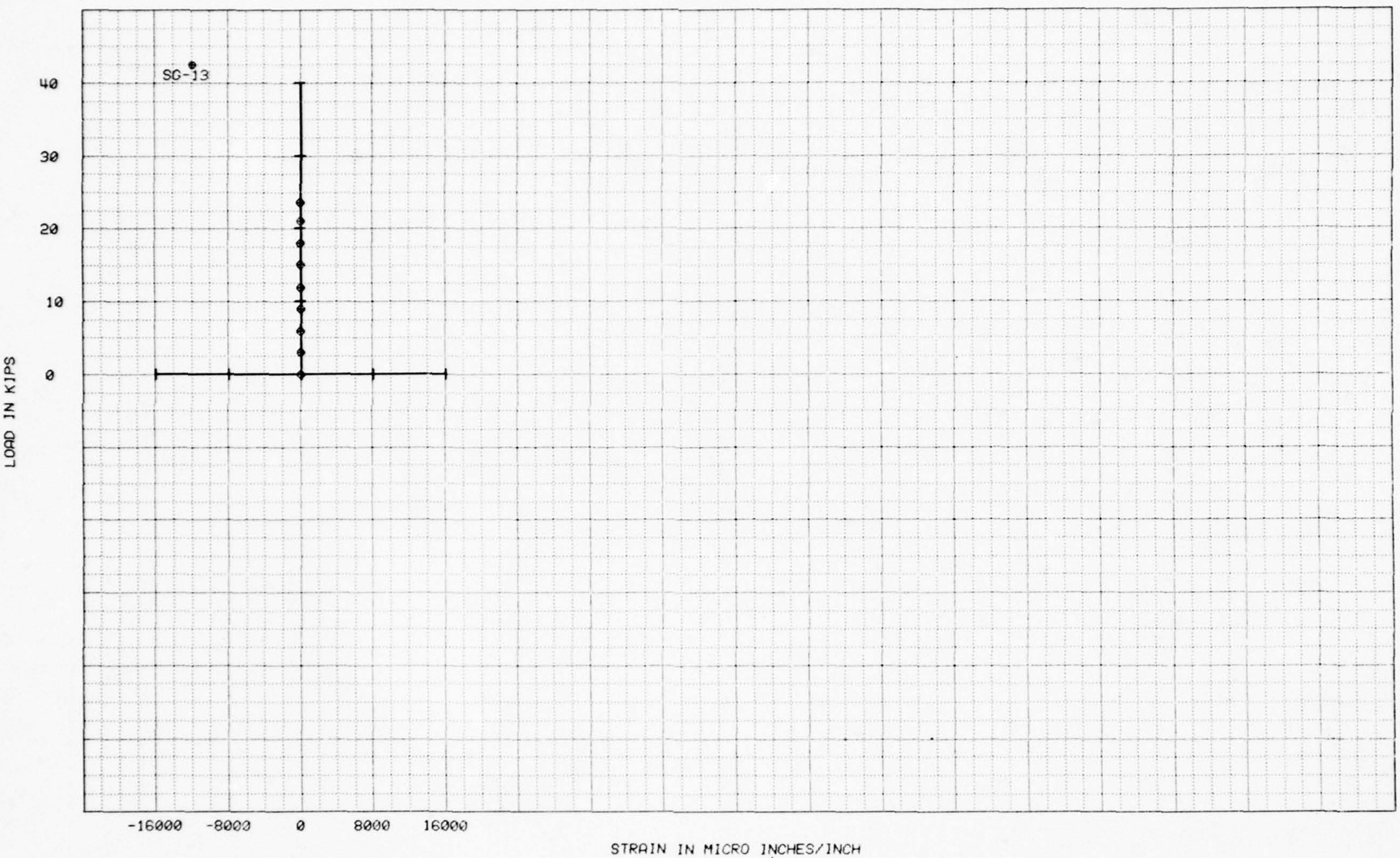


FIGURE B-1 Continued Strain Gage Data 75T060105-1001
3/16 Laminate 1.0 Dia Damage Hole

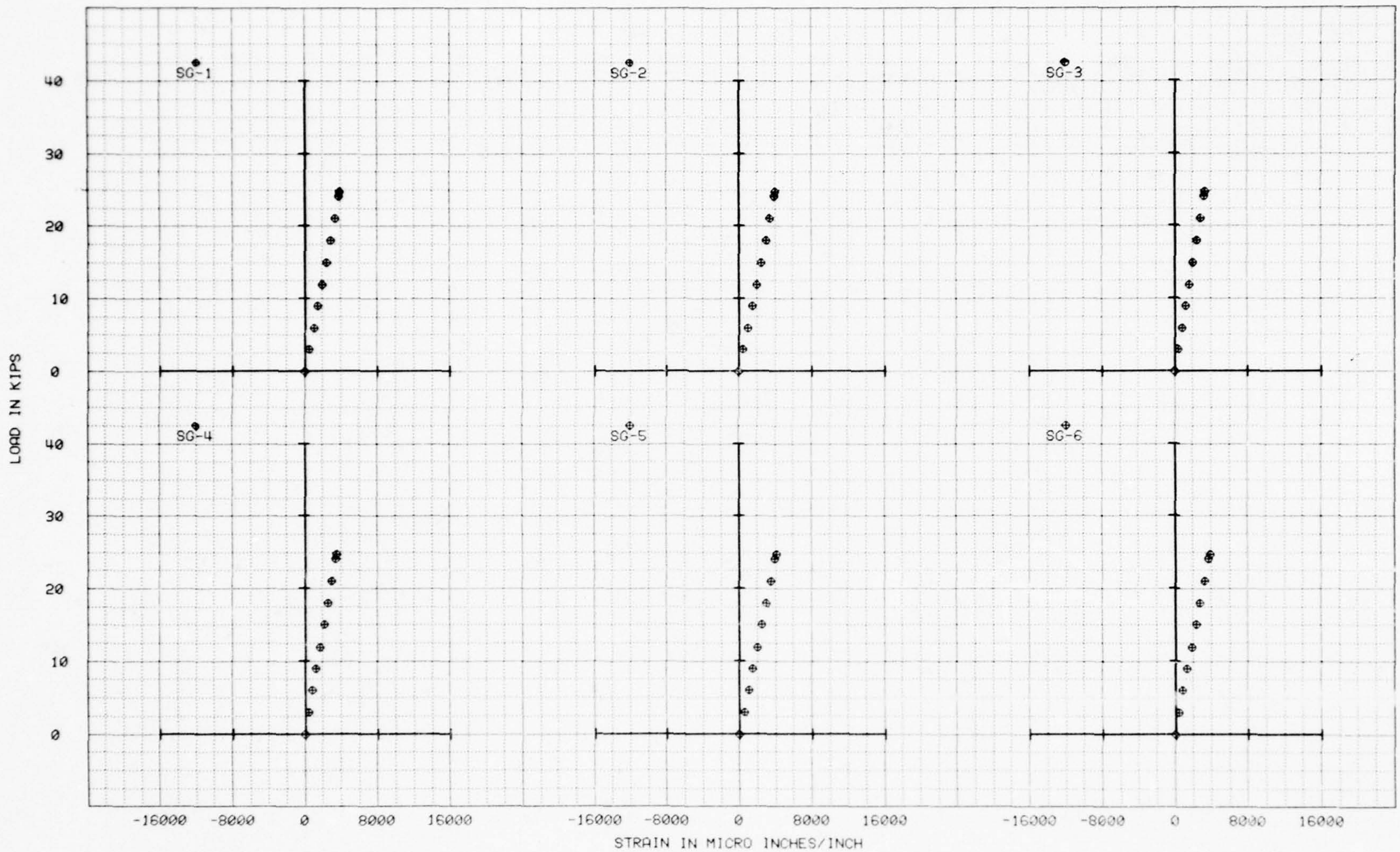


FIGURE B-2 Strain Gage Data 75T060105-1003
3/16 Laminate 1.0 Dia Damage Hole

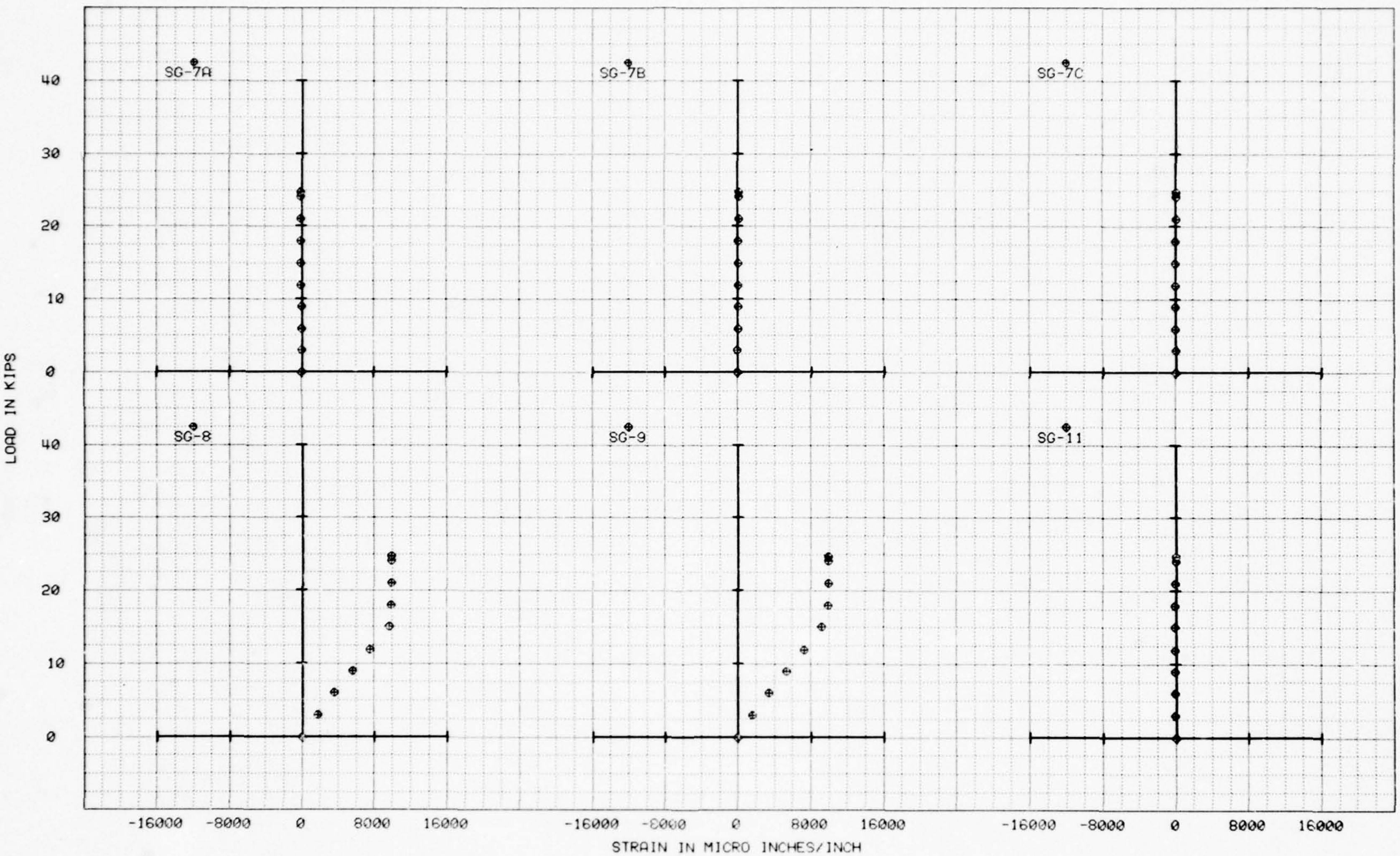


FIGURE B-2 Continued Strain Gage Data 75T060105-1003
3/16 Laminate 1.0 Dia Damage Hole

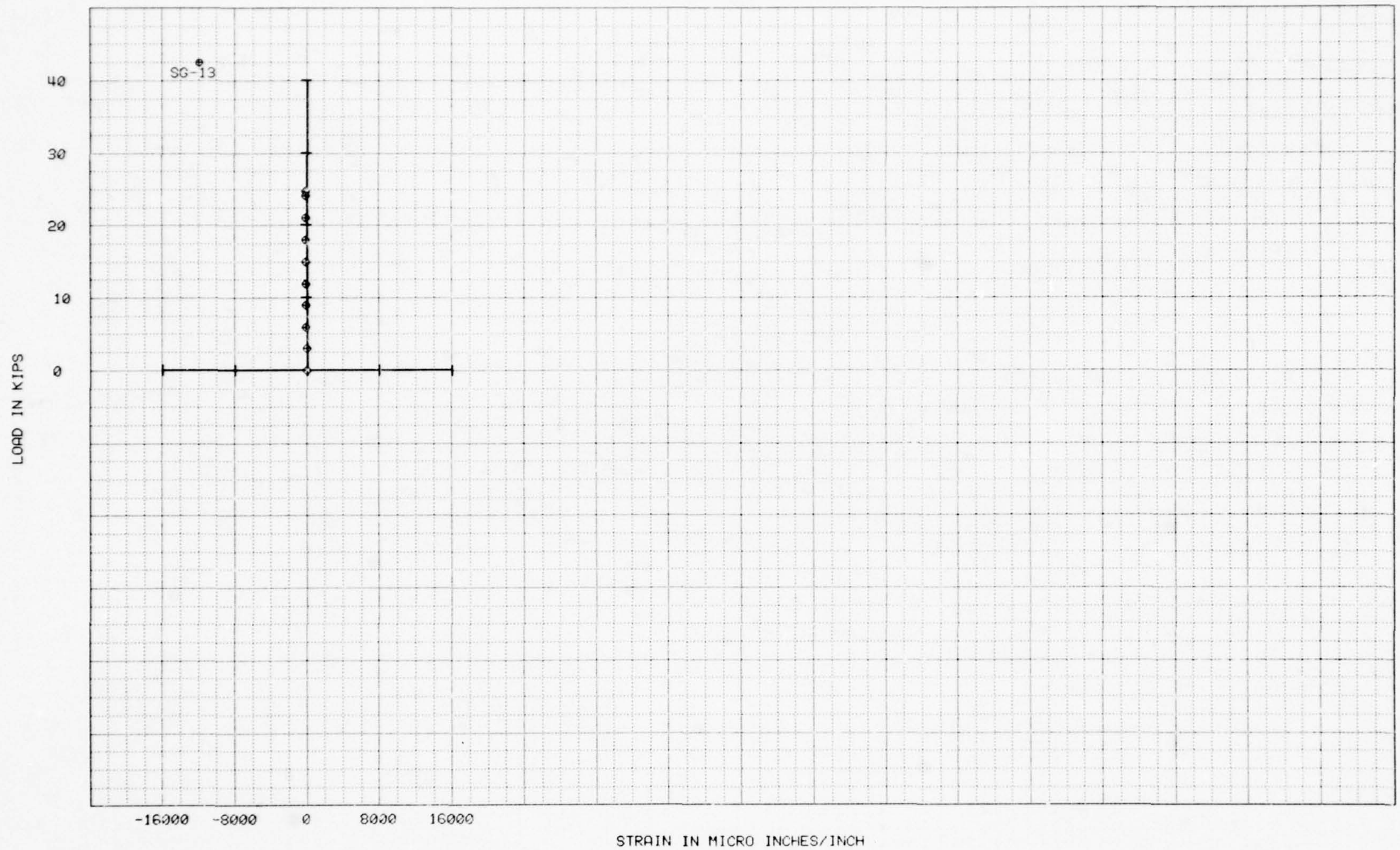


FIGURE B-2 Continued Strain Gage Data 75T060105-1003
3/16 Laminate 1.0 Dia Damage Hole

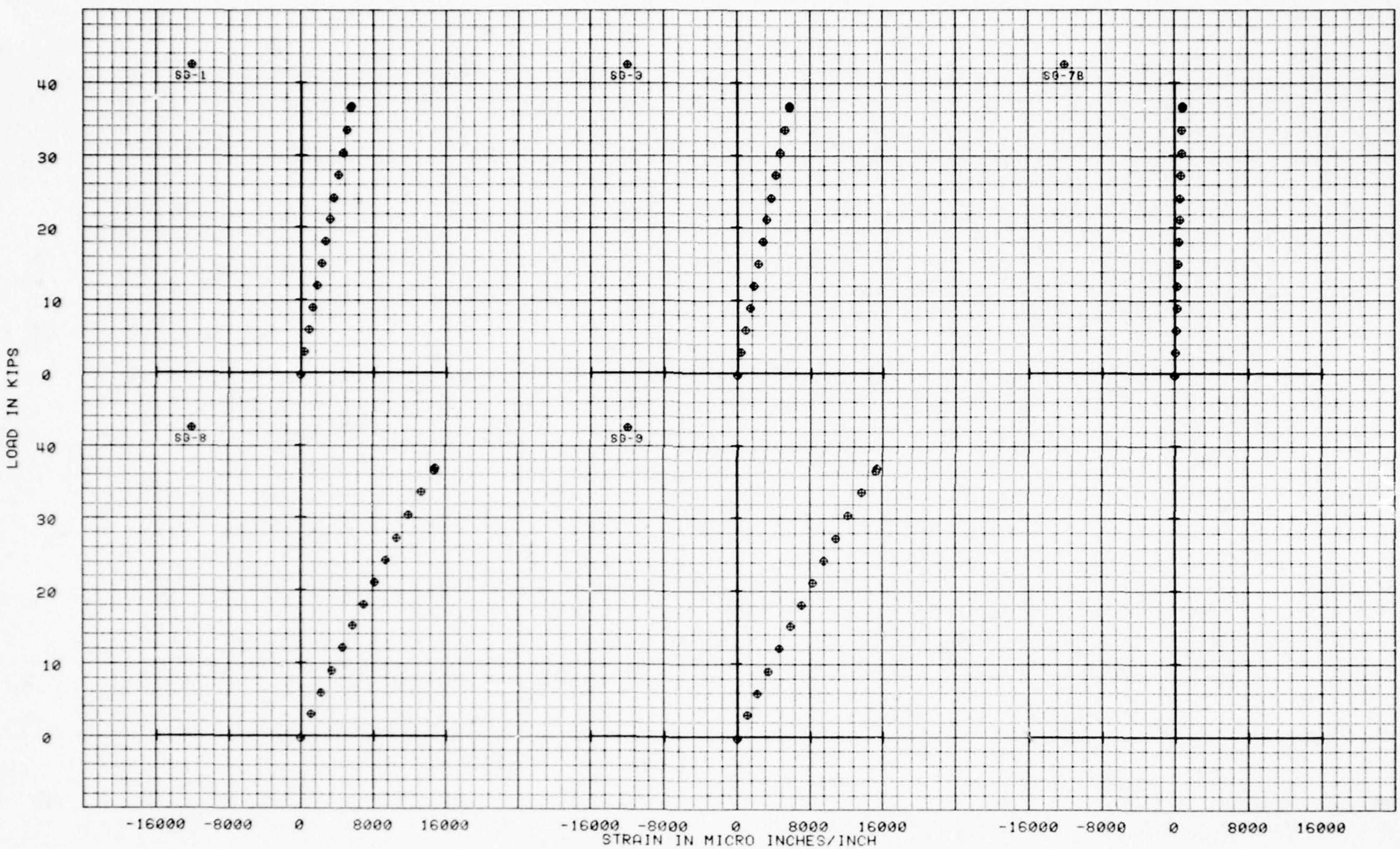


FIGURE B-3 Strain Gage Data 75T060105-1019 No. 1
3/16 Laminate 1.0 Dia Damage Hole

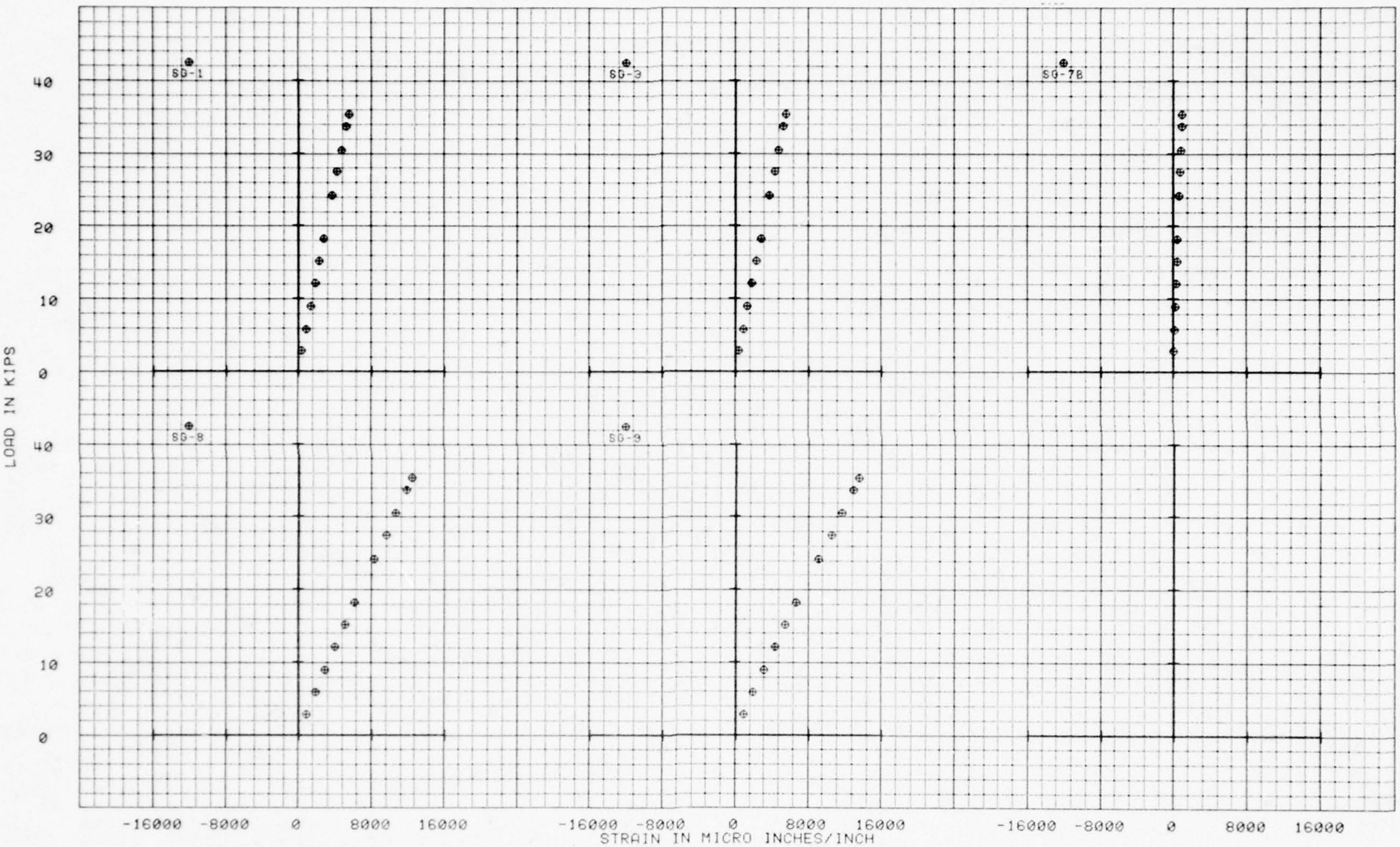


FIGURE B-4 STRAIN GAGE DATA 75T060105-1019 No. 2
3/16 Laminate 1.0 Dia Damage Hole

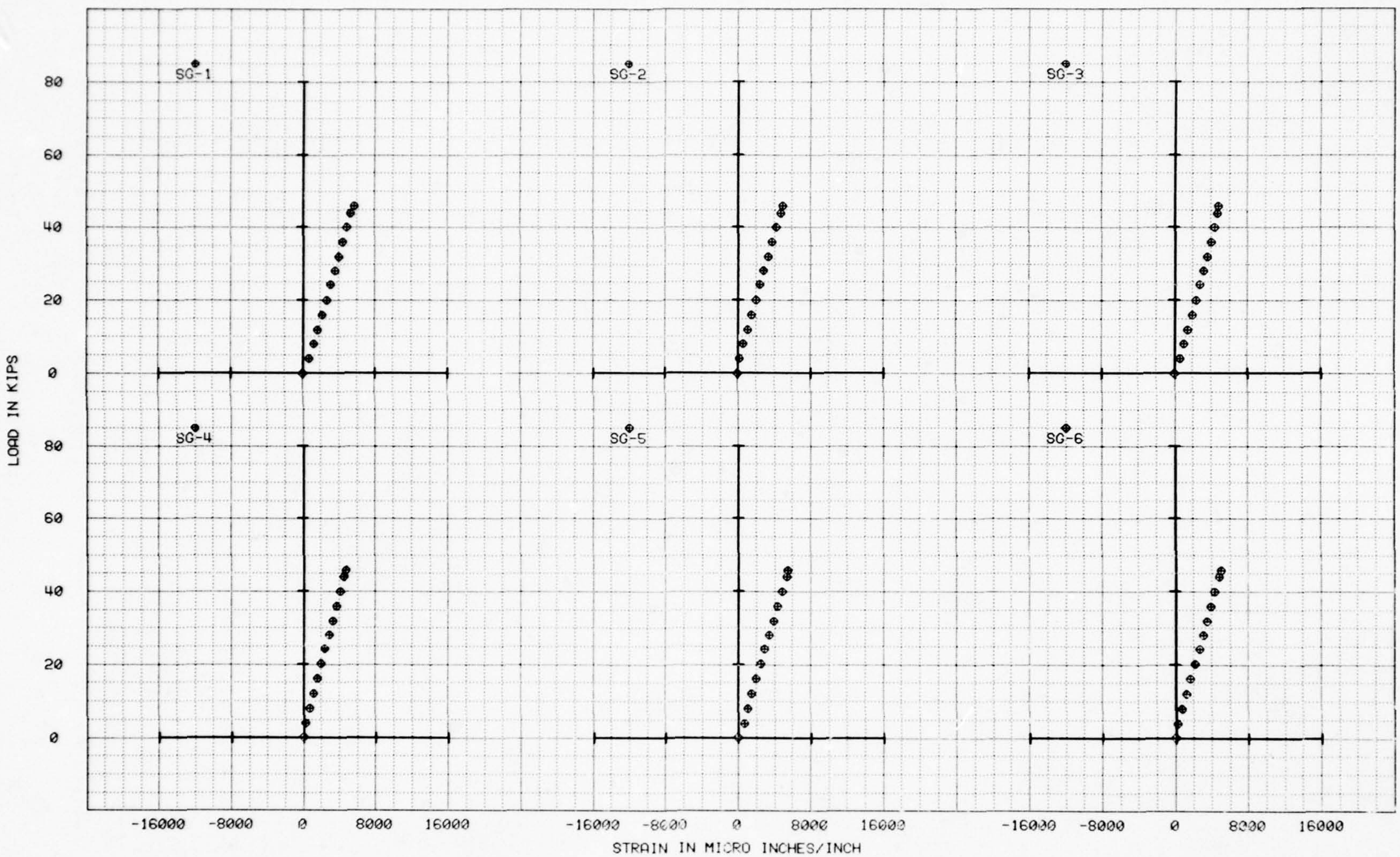


FIGURE B-5 Strain Gage Data 75T060105-1005 No. 1
3/16 Laminate 2.5 Dia Damage Hole

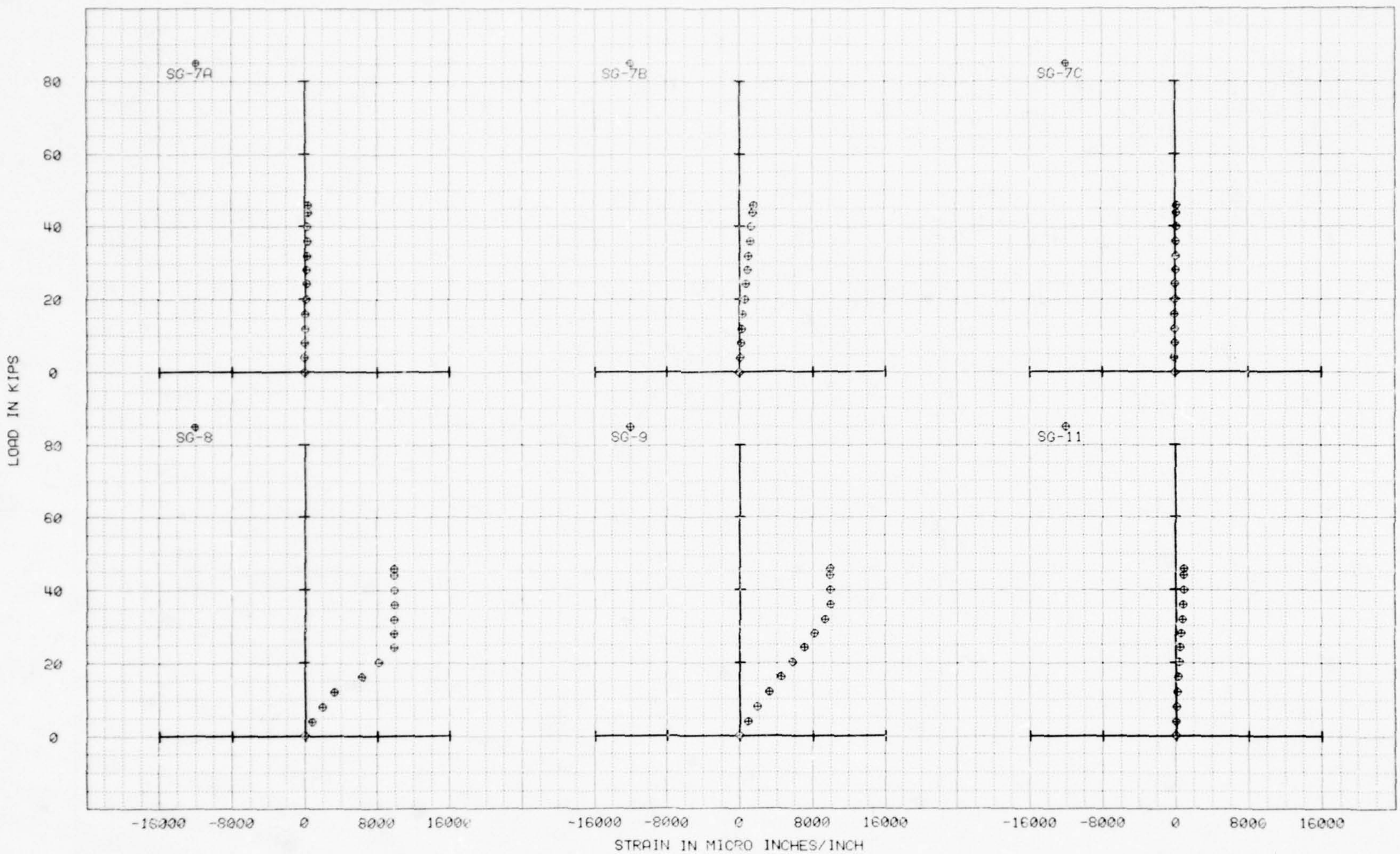


FIGURE B-5 Continued Strain Gage Data 75T060105-1005 No. 1
3/16 Laminate 2.5 Dia Damage Hole

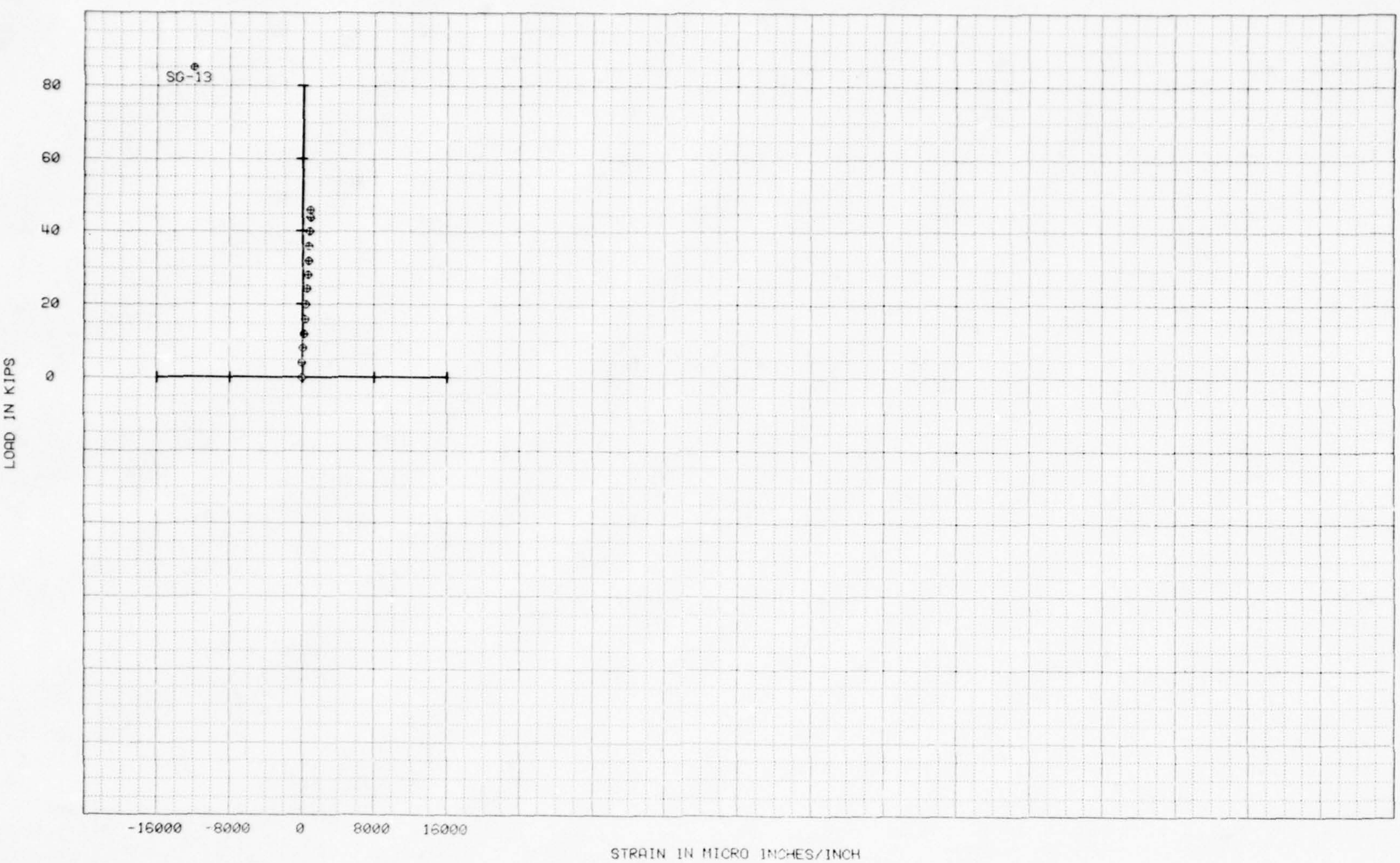


FIGURE B-5 Continued Strain Gage Data 75T060105-1005 No. 1
3/16 Laminate 2.5 Dia Damage Hole

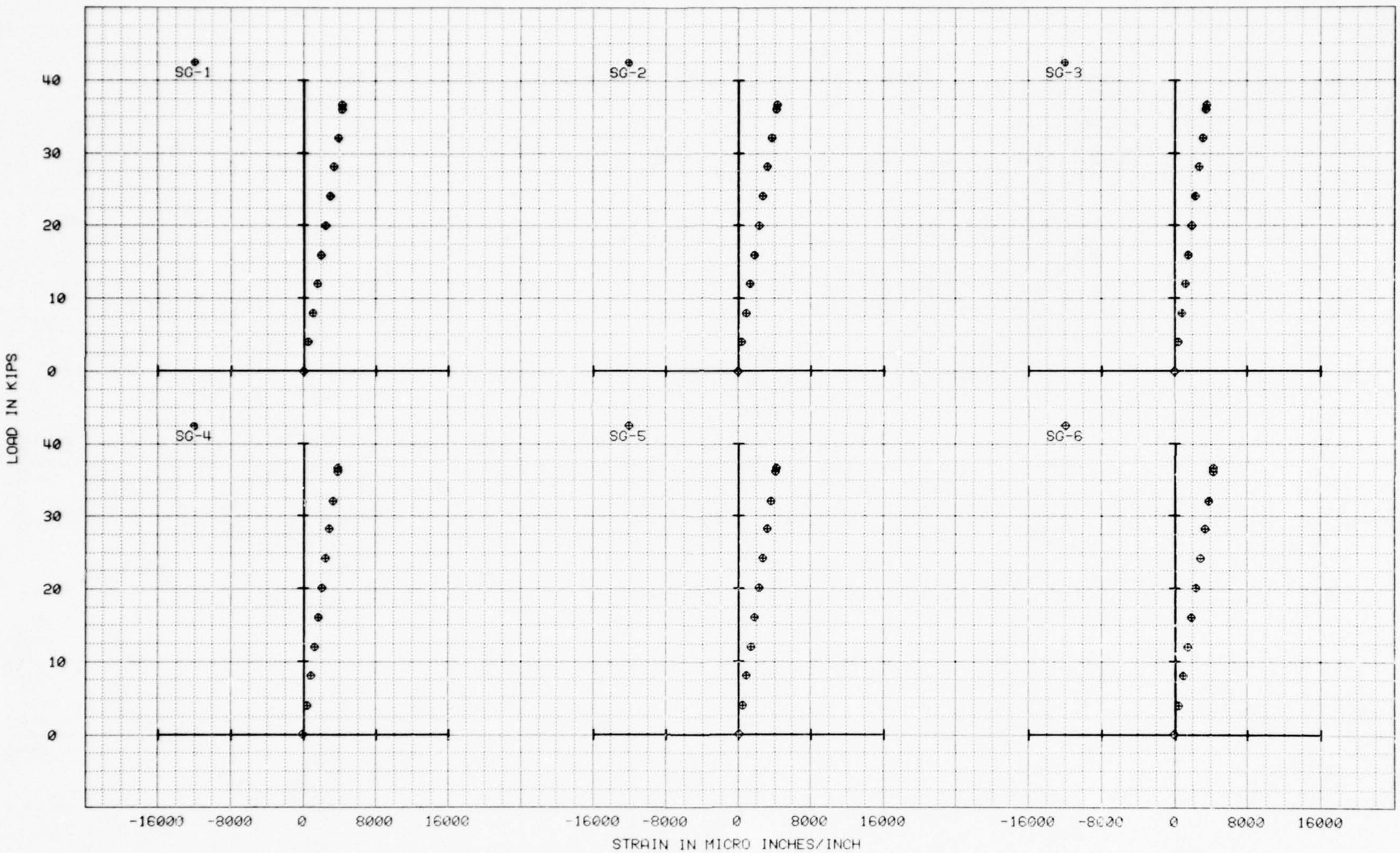


FIGURE B-6 Strain Gage Data 75T060105-1007
3/16 Laminate 2.5 Dia Damage Hole

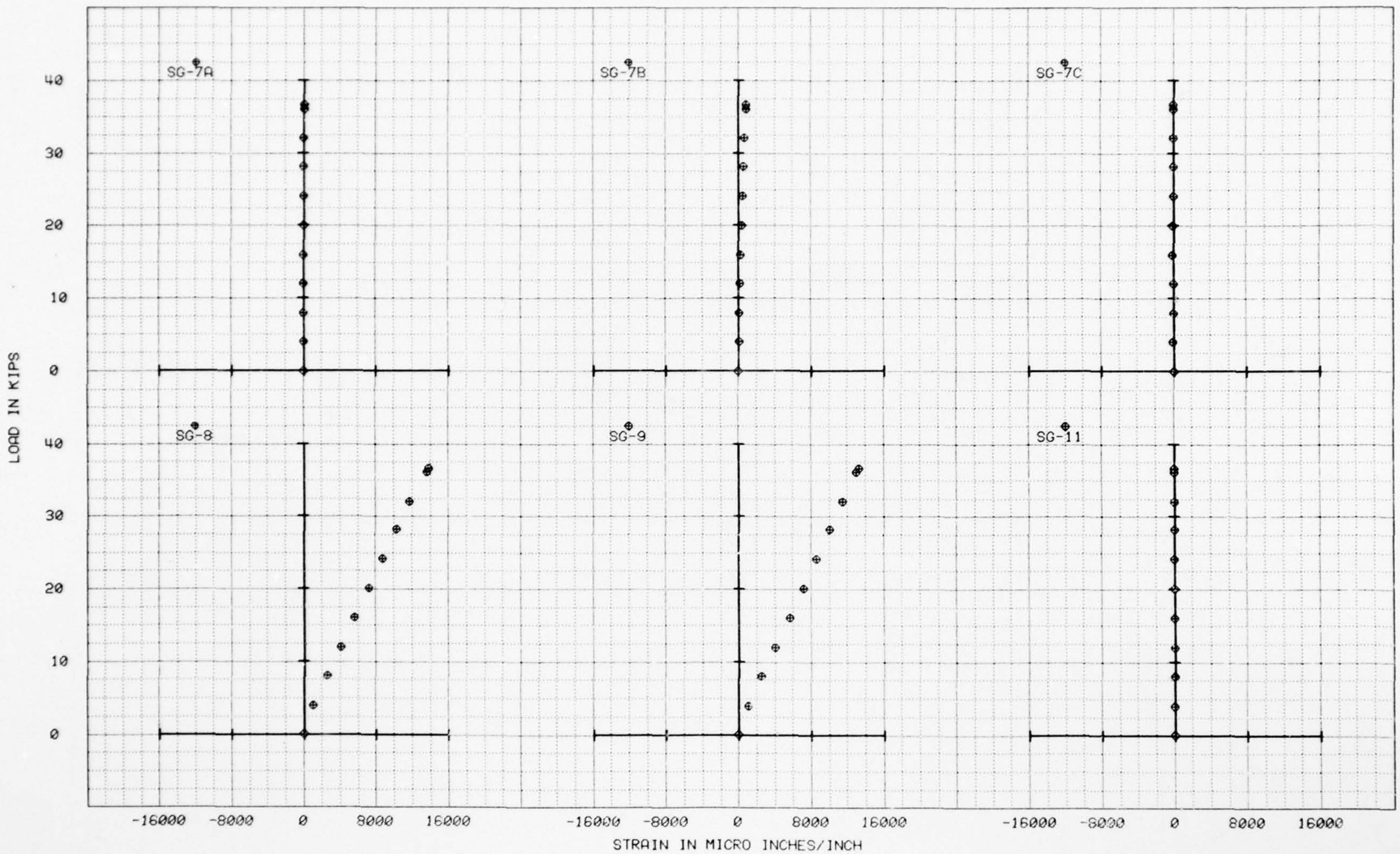


FIGURE B-6 Continued Strain Gage Data 75T060105-1007
3/16 Laminate 2.5 Dia Damage Hole

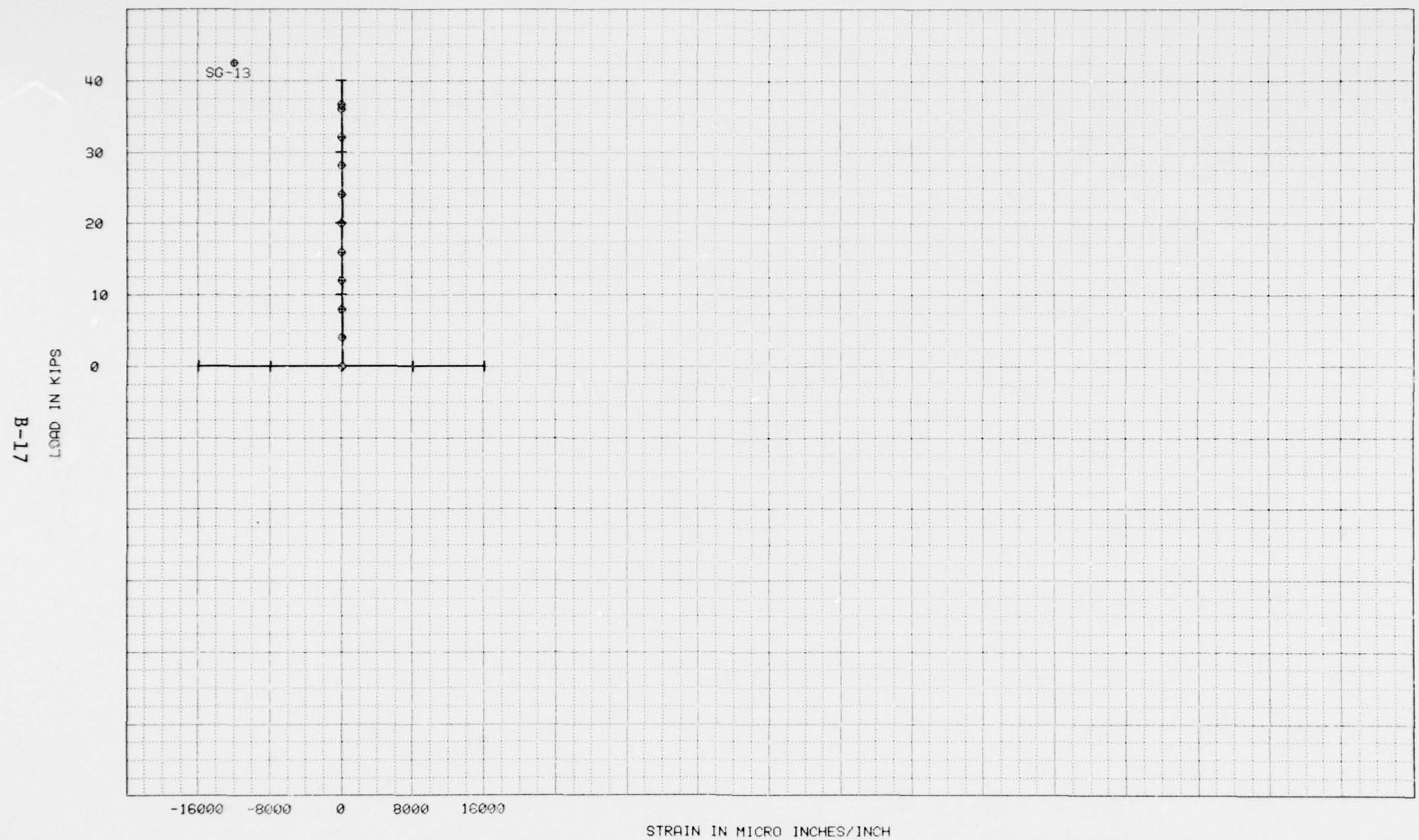


FIGURE B-6 Continued Strain Gage Data 75T060105-1007
3/16 Laminate 2.5 Dia Damage Hole

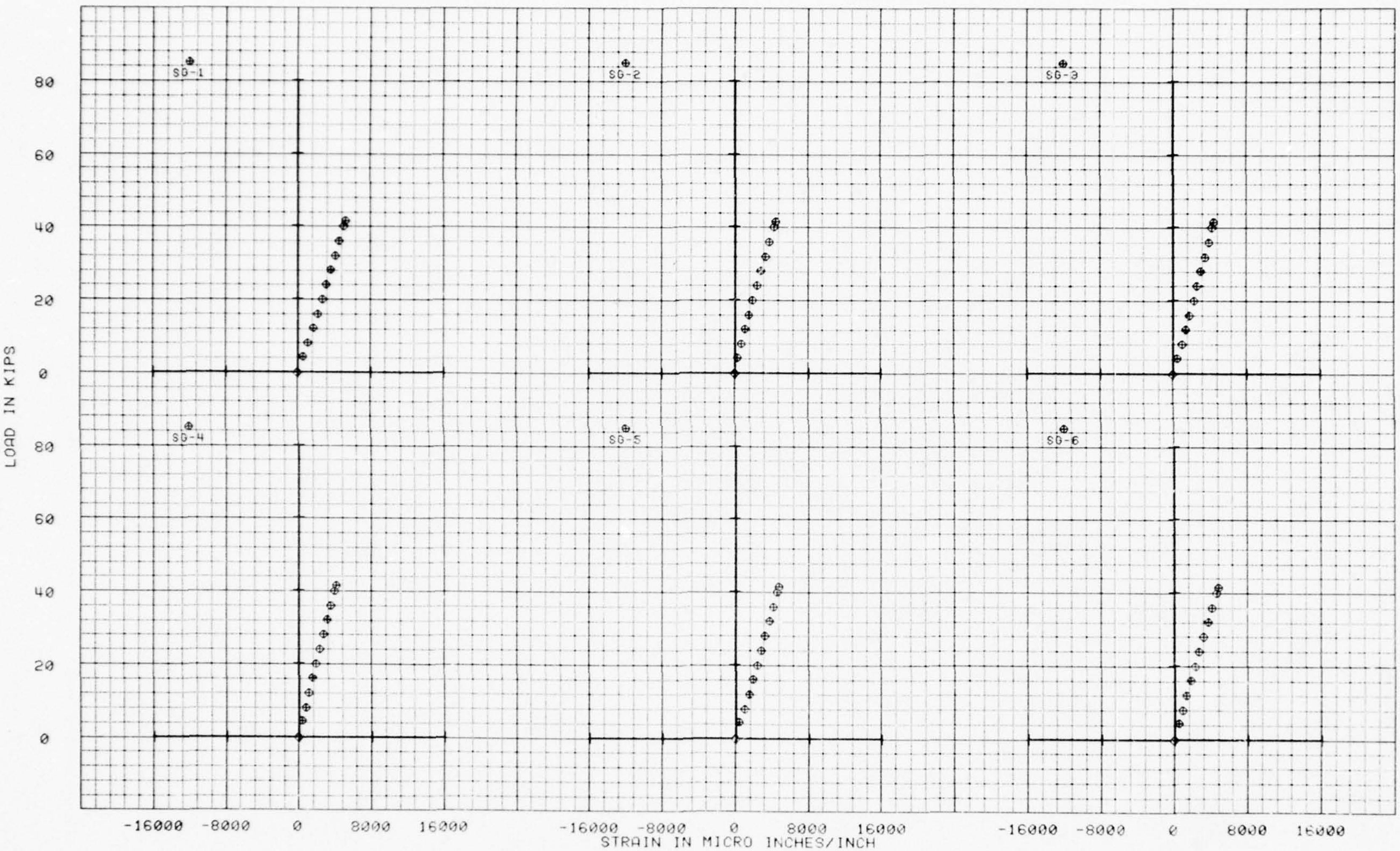


FIGURE B-7 Strain Gage Data 75T060105-1005 No. 2
3/16 Laminate 2.5 Dia Damage Hole

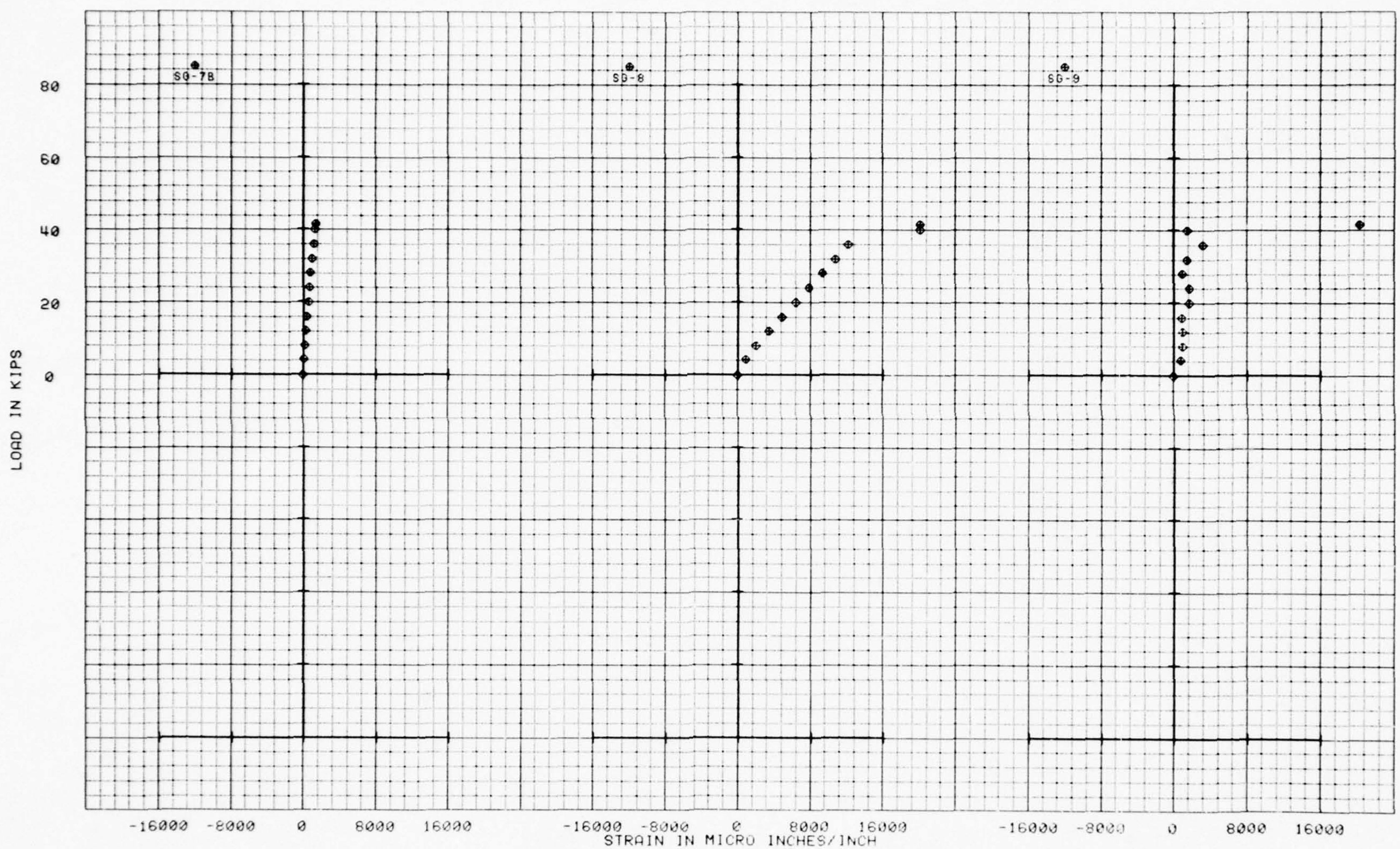


FIGURE B-7 Continued Strain Gage Data 75T060105-1005 No. 2
3/16 Laminate 2.5 Dia Damage Hole

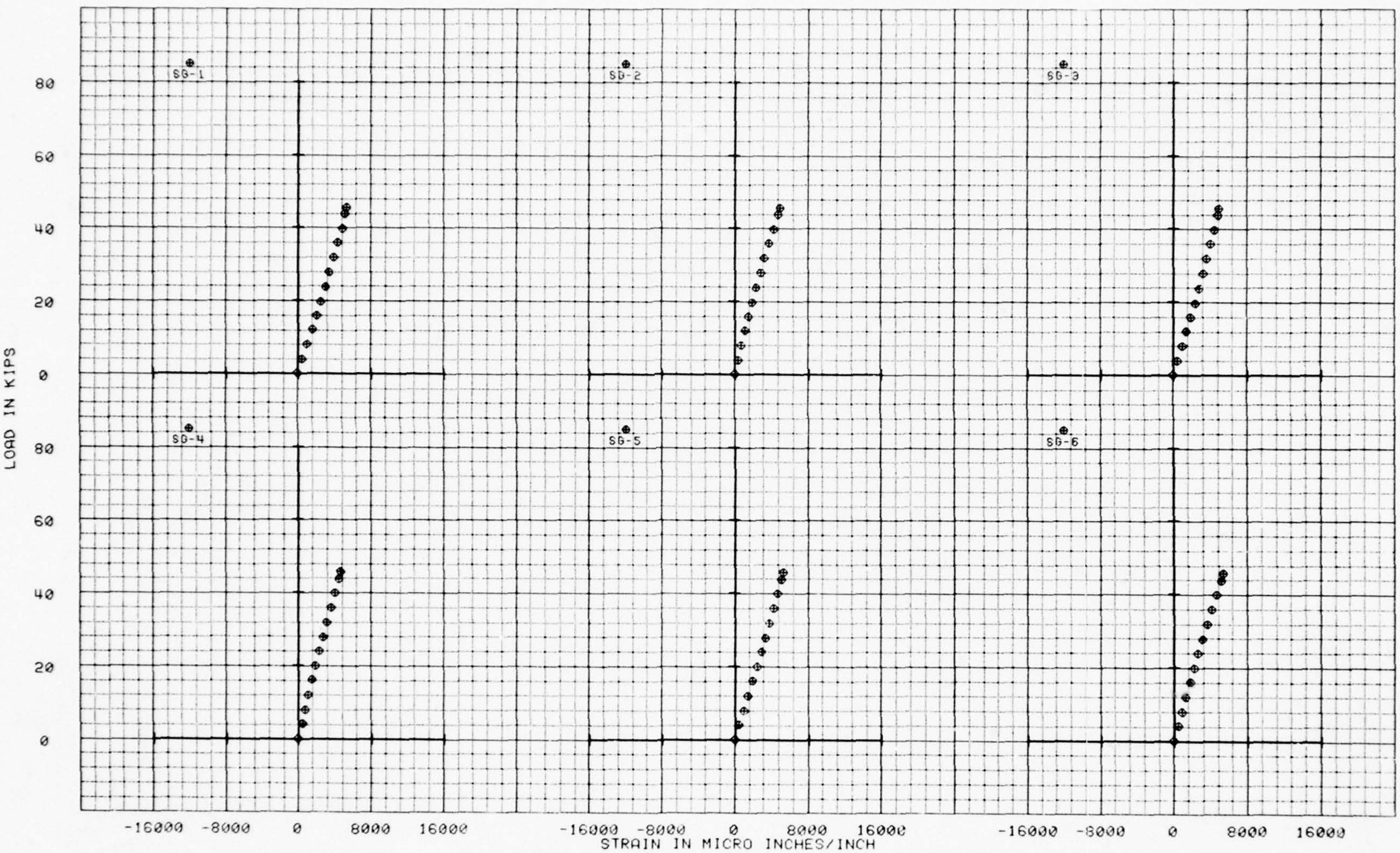


FIGURE B-8 Strain Gage Data 75T060105-1005 No. 3
3/16 Laminate 2.5 Dia Damage Hole

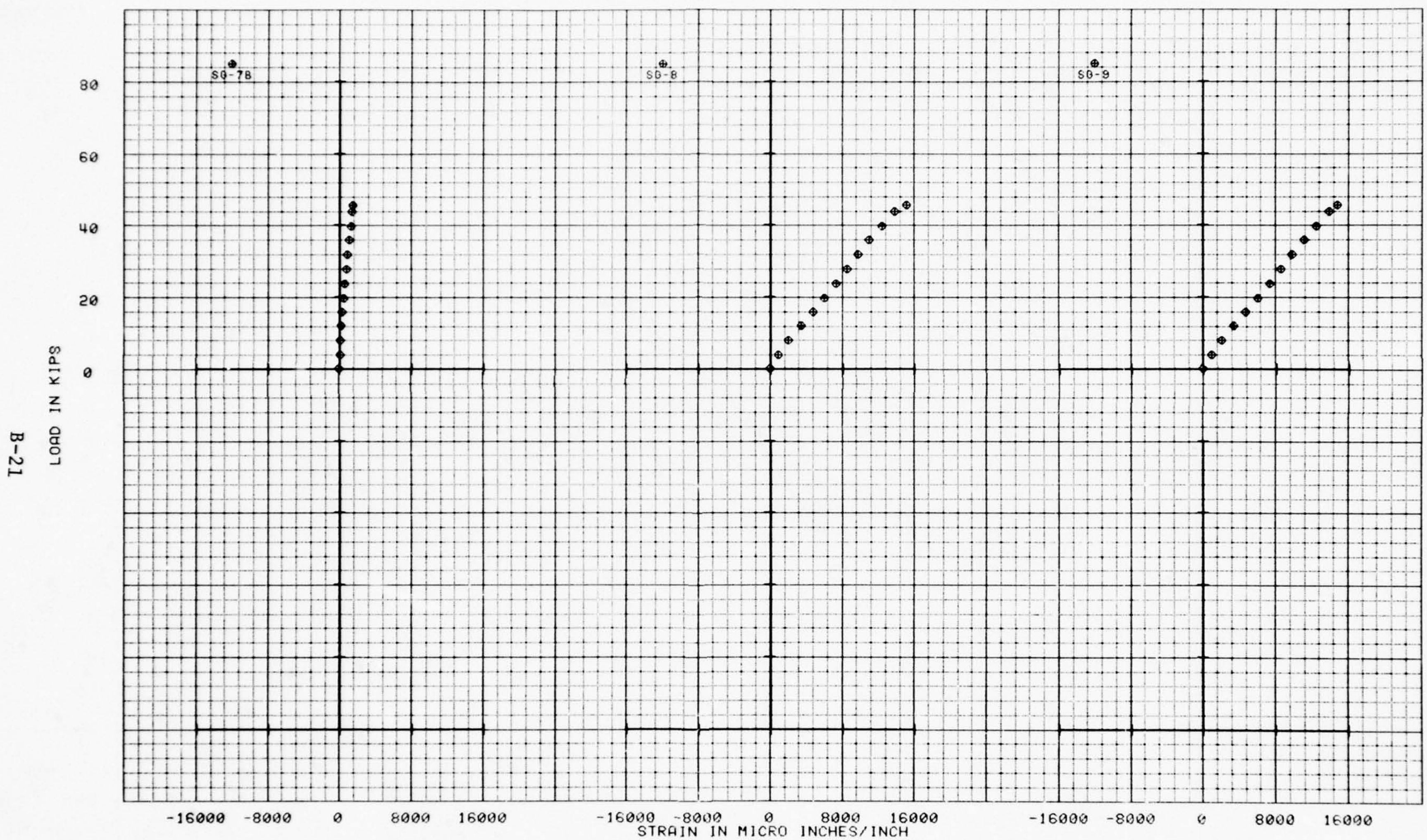


FIGURE B-8 Continued Strain Gage Data 75T060105-1005 No. 3
3/16 Laminate 2.5 Dia Damage Hole

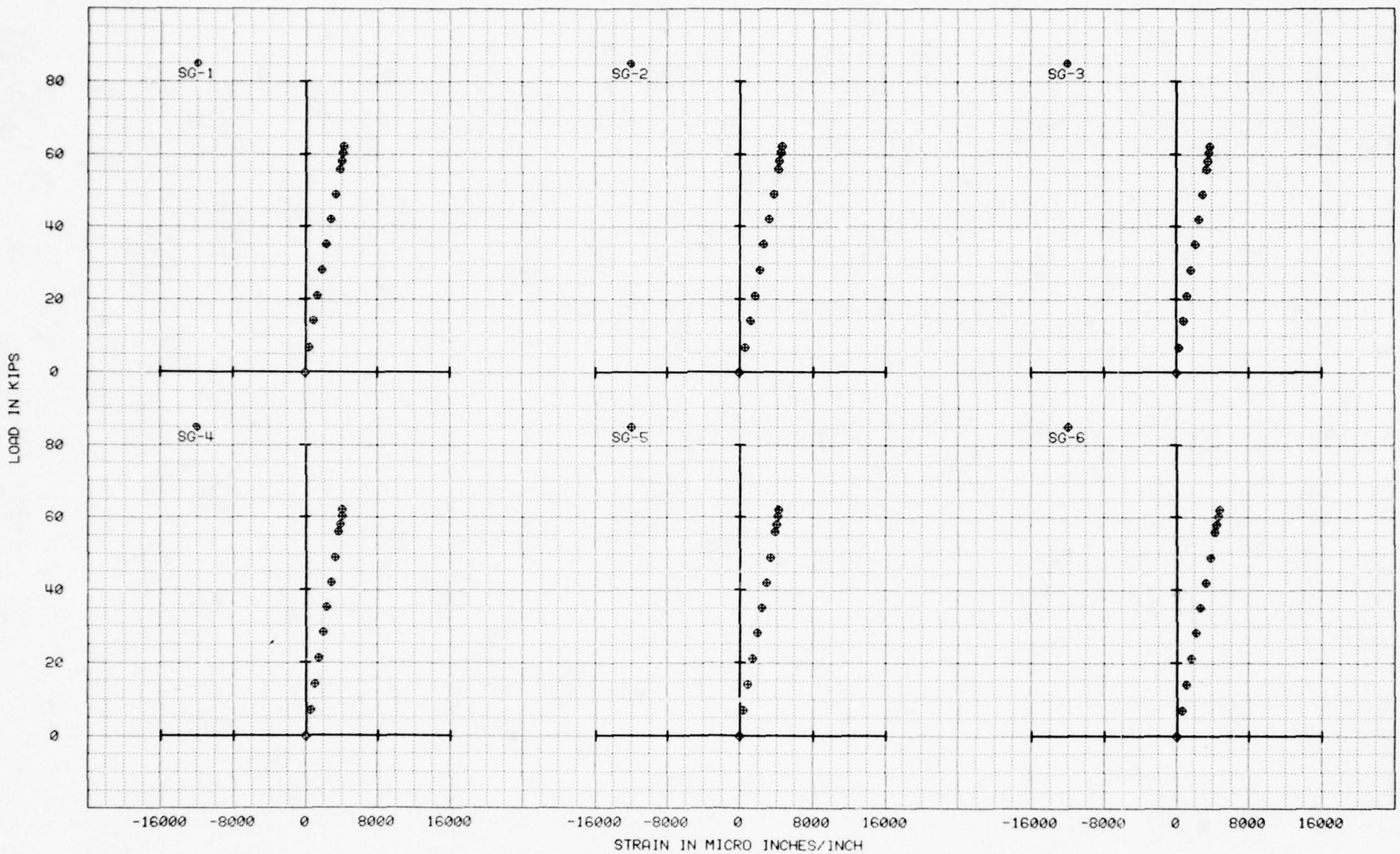


FIGURE B-9 Strain Gage Data 75T060105-1011
3/16 Laminate 4.0 Dia Damage Hole

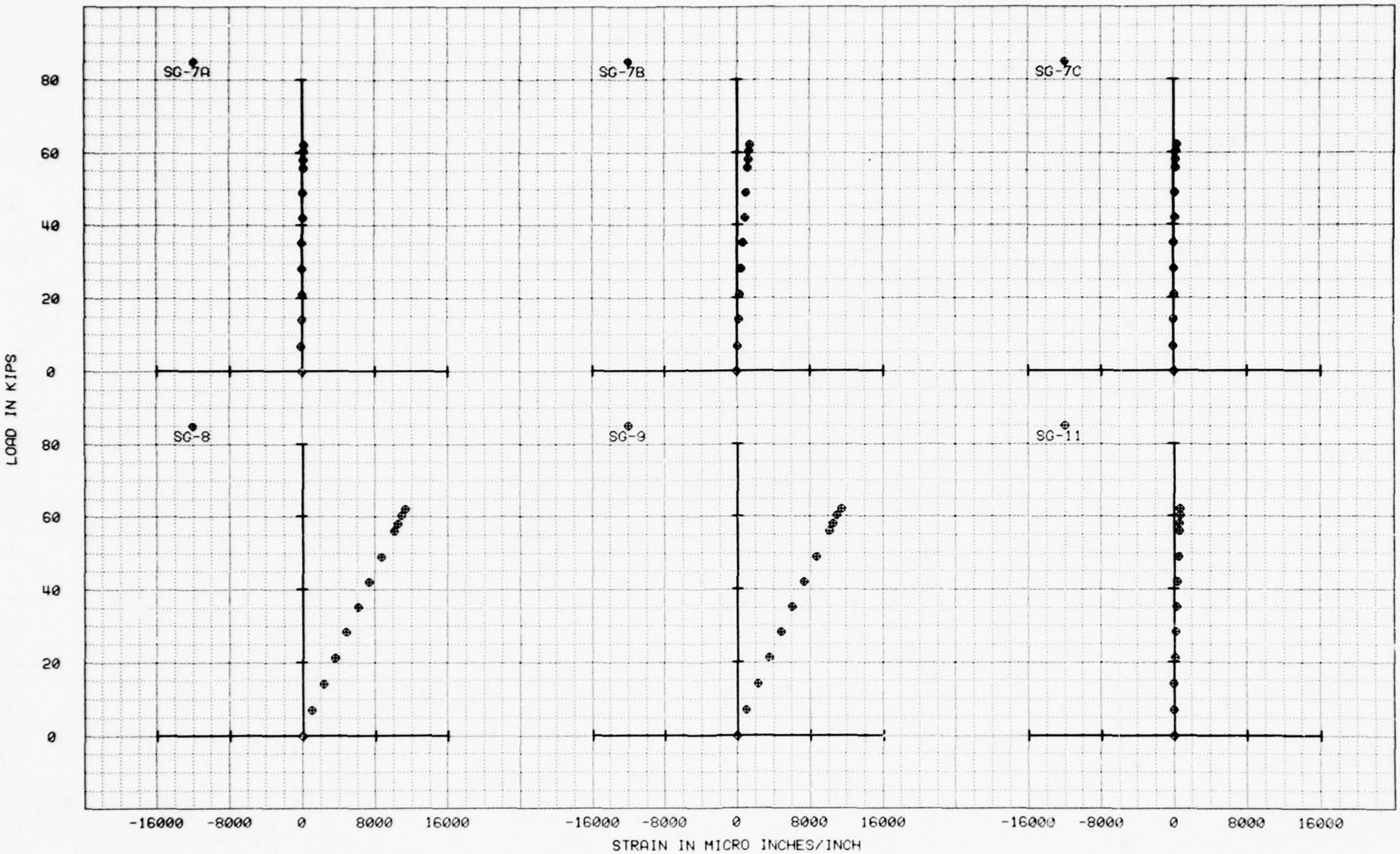


FIGURE B-9 Continued Strain Gage Data 75T060105-1011
3/16 Laminate 4.0 Dia Damage Hole

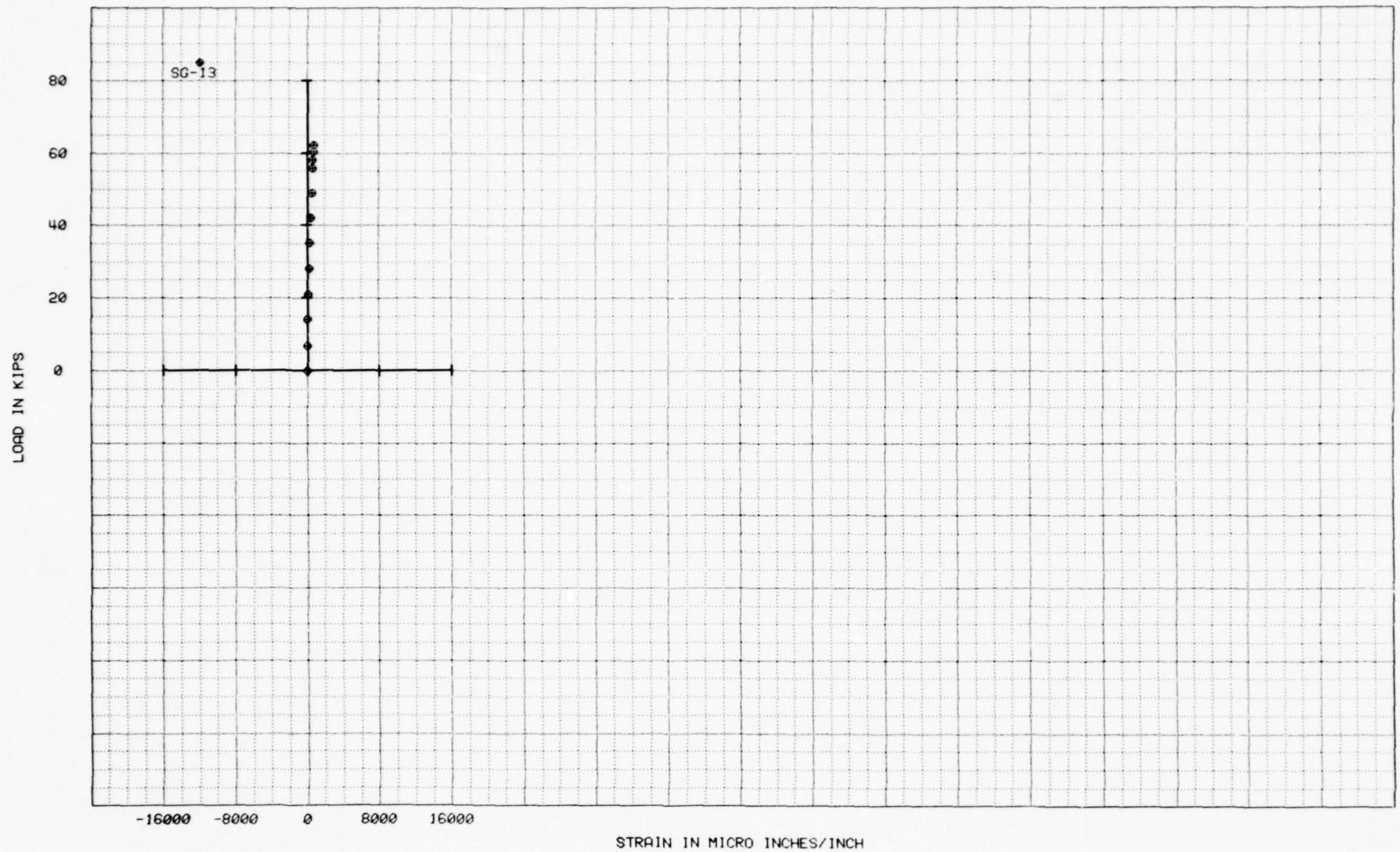


FIGURE B-9 Continued Strain Gage Data 75T060105-1011
3/16 Laminate 4.0 Dia Damage Hole

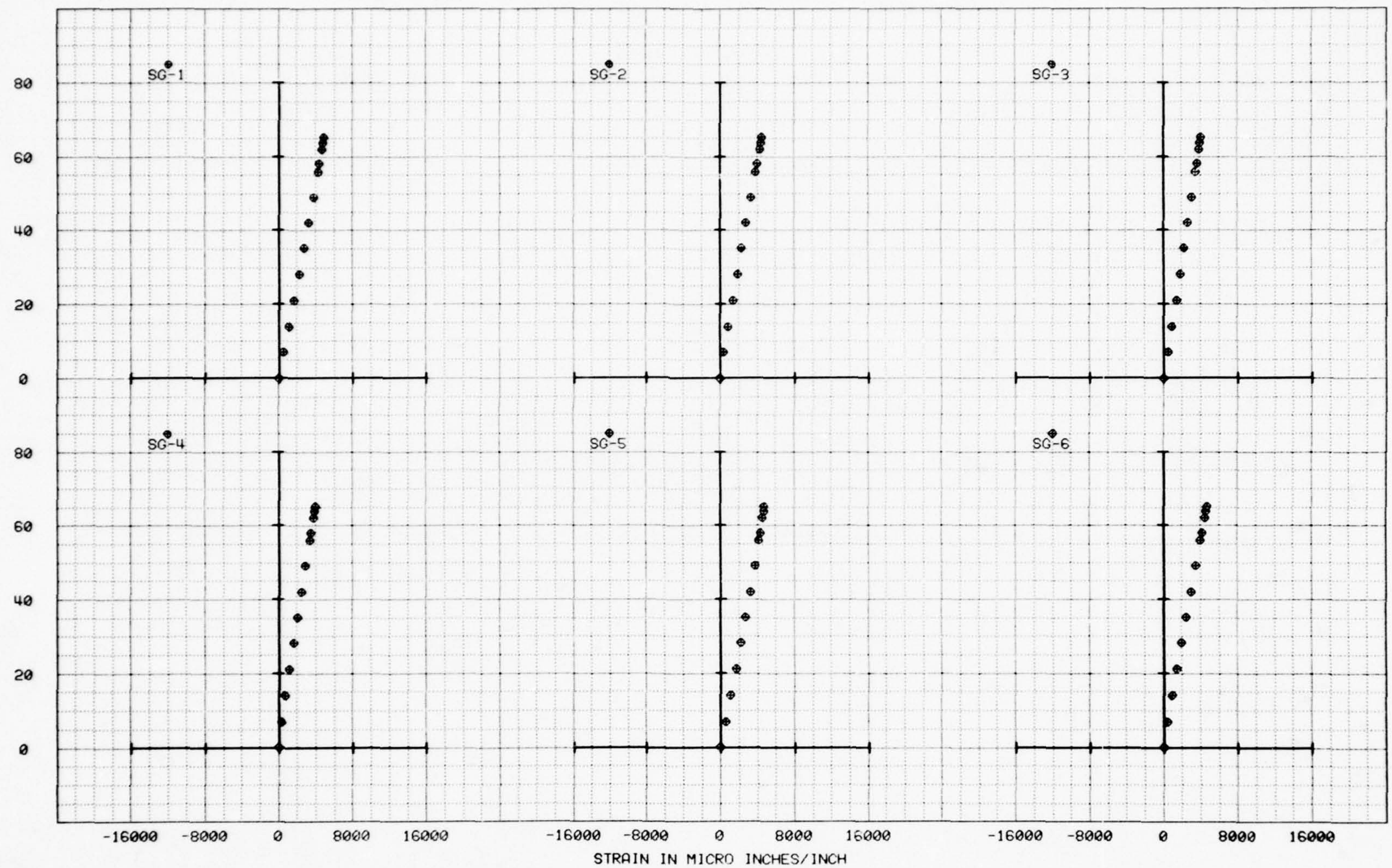


FIGURE B-10 Strain Gage Data 75T060105-1013
3/16 Laminate 4.0 Dia Damage Hole

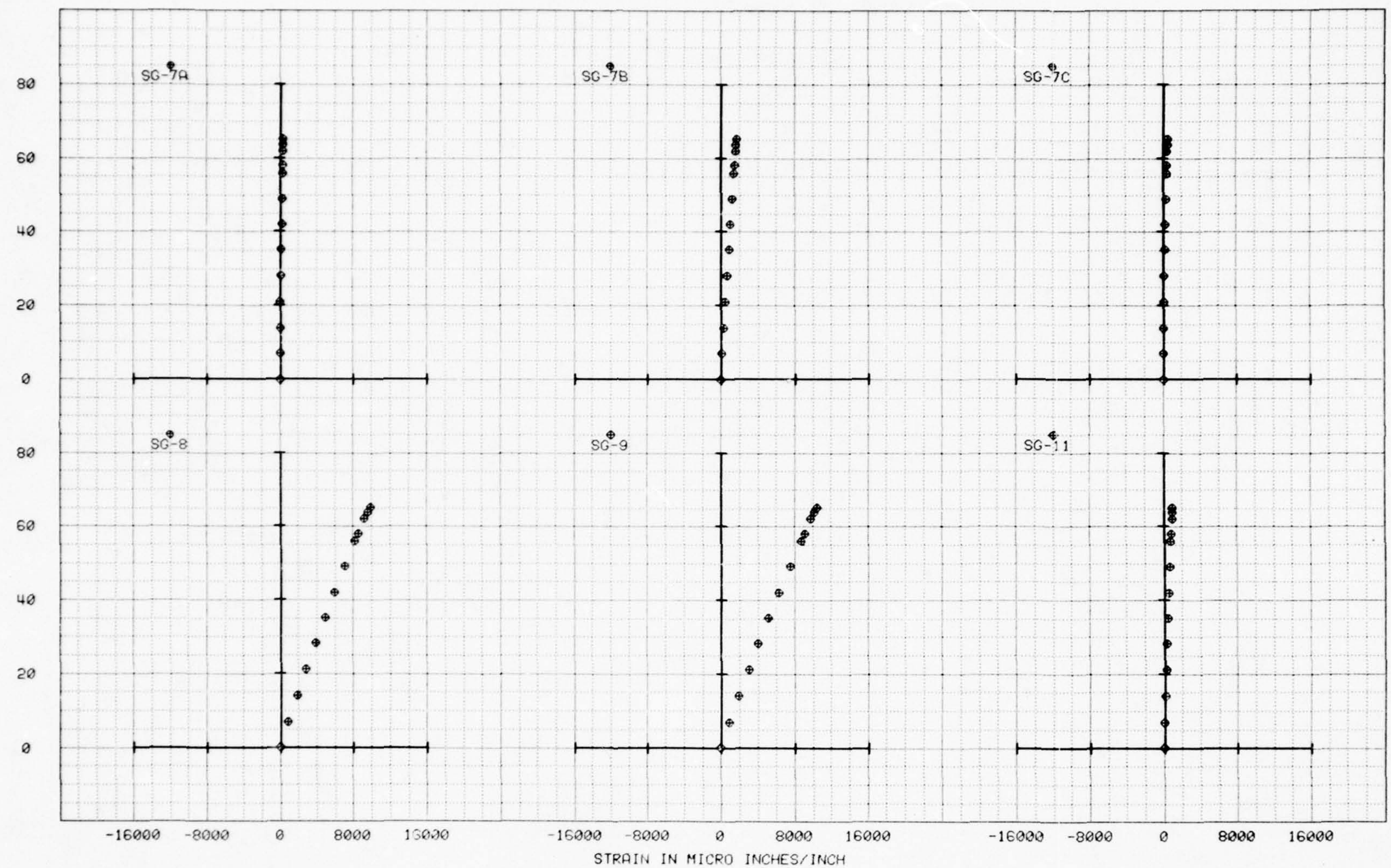


FIGURE B-10 Continued Strain Gage Data 75T060105-1013
3/16 Laminate 4.0 Dia Damage Hole

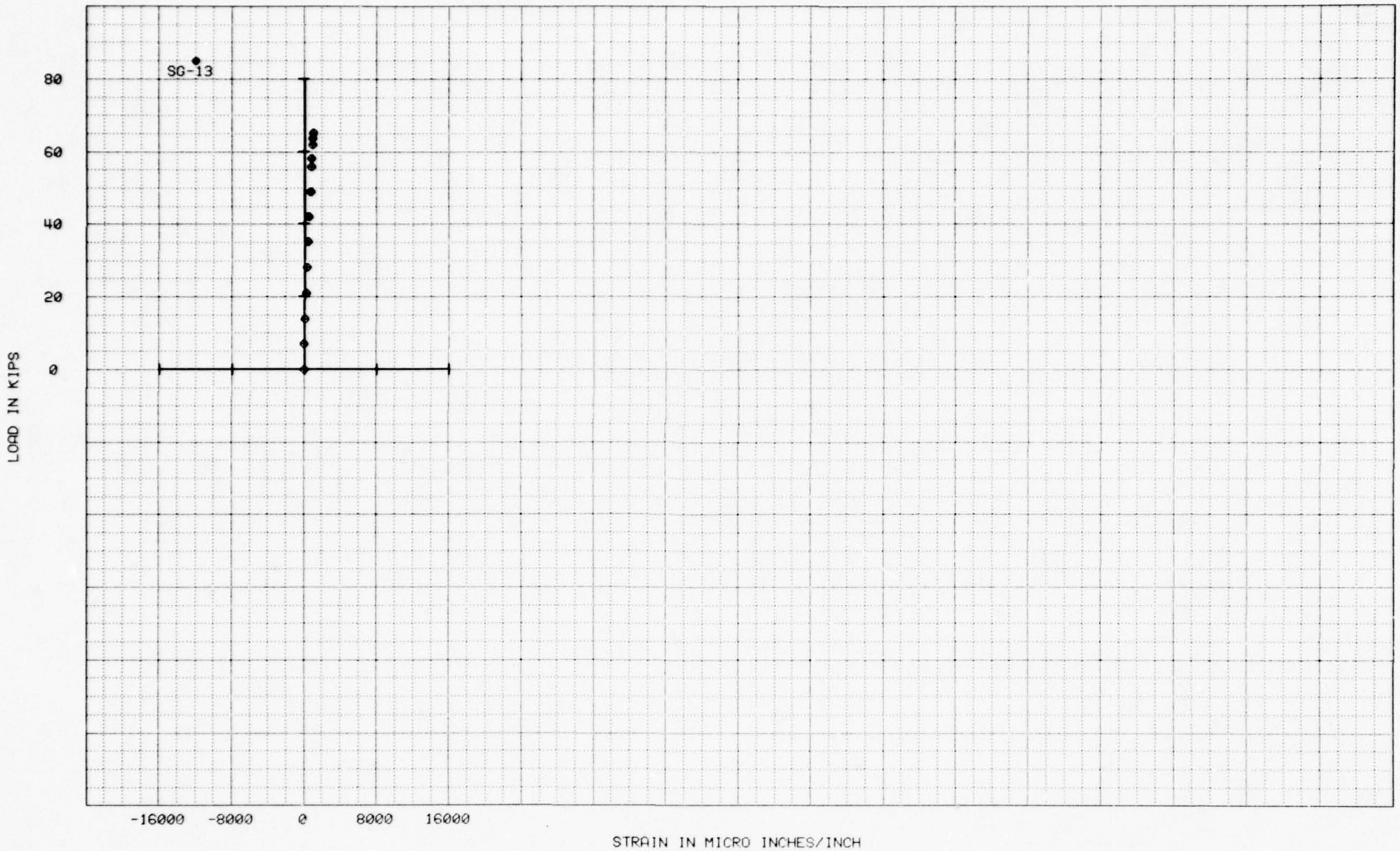


FIGURE B-10 Continued Strain Gage Data 75T060105-1013
3/16 Laminate 4.0 Dia Damage Hole

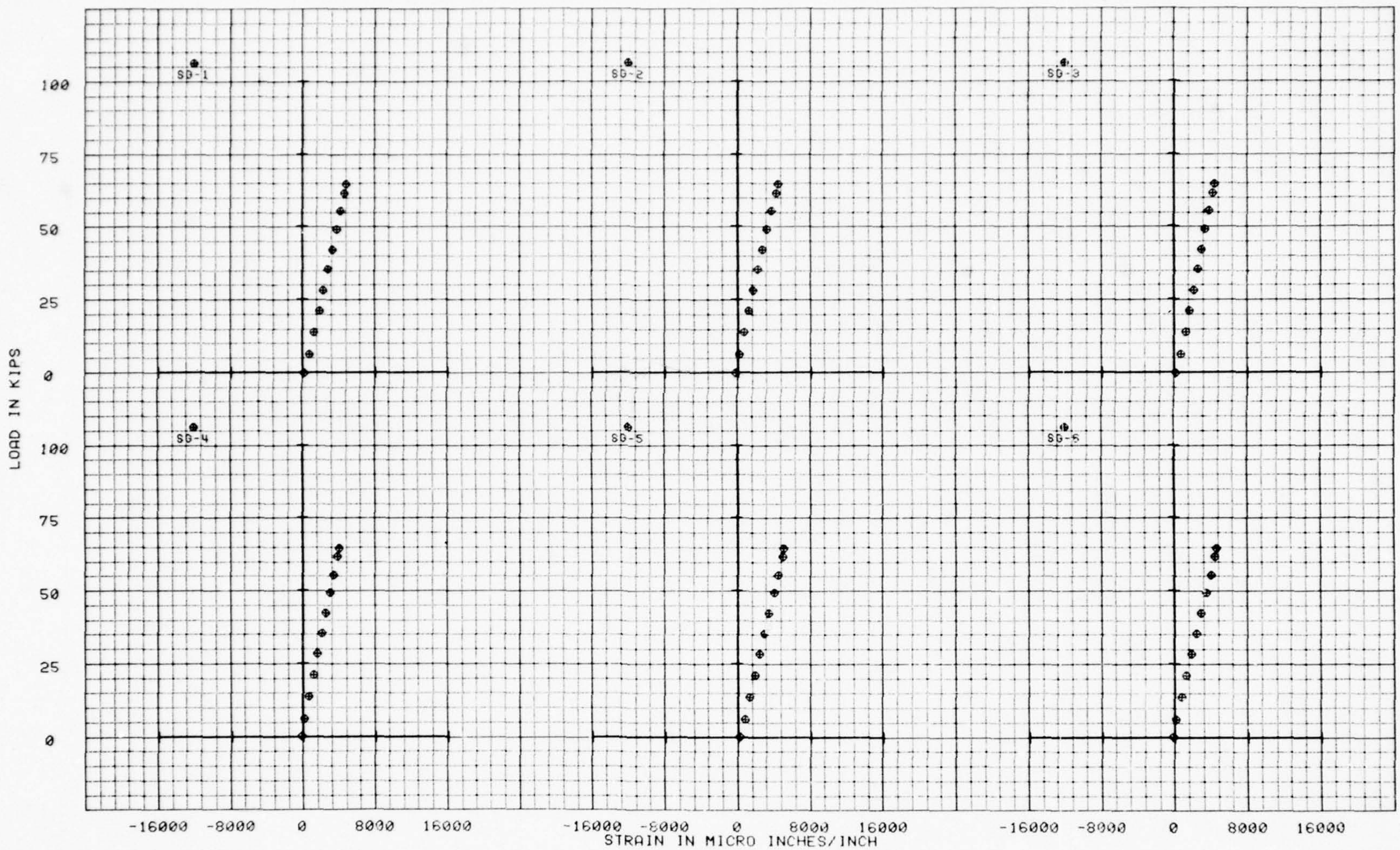


FIGURE B-11 Strain Gage Data 75T060105-1015 No. 1
3/16 Laminate 4.0 Dia Damage Hole

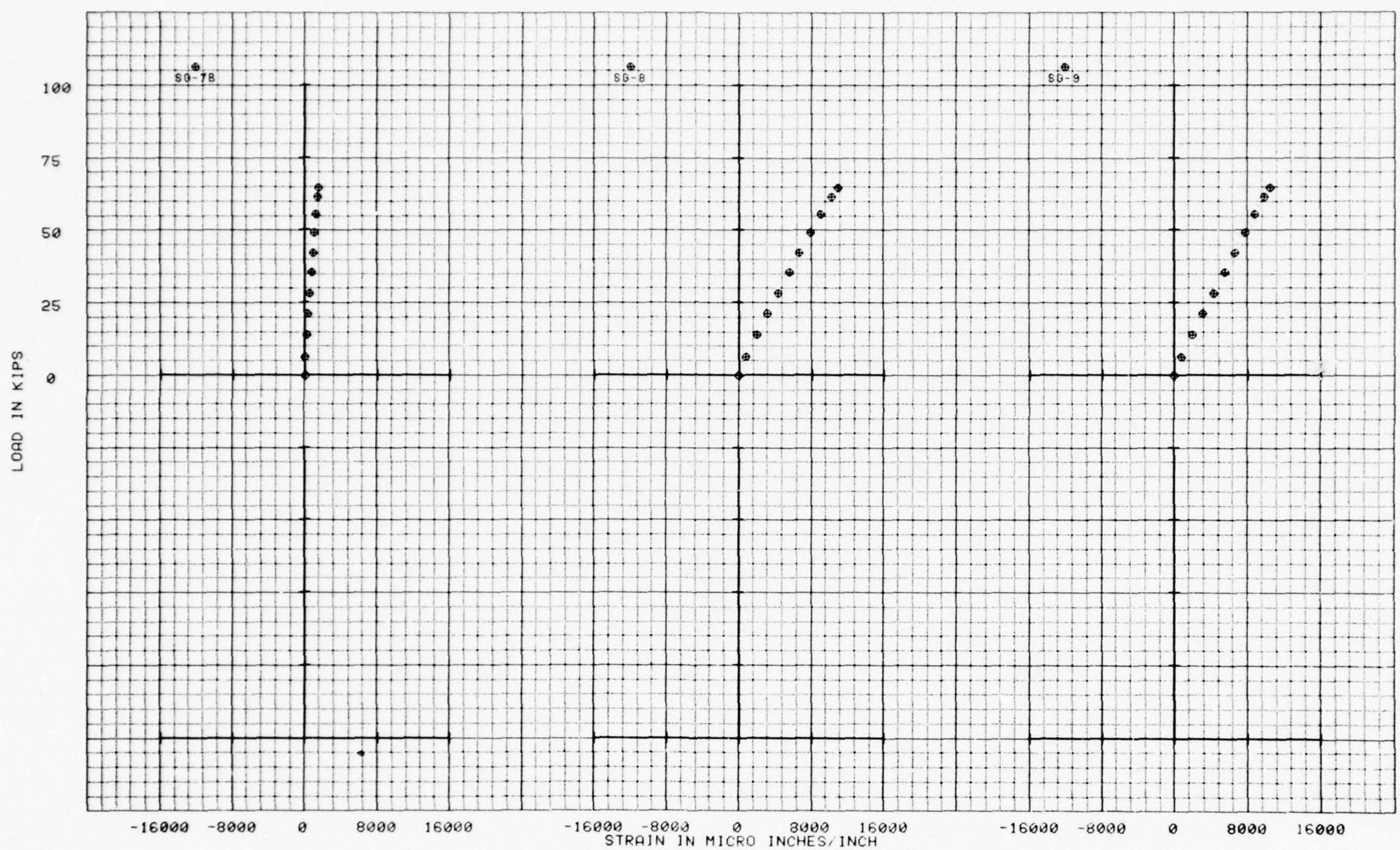


FIGURE B-11 Continued Strain Gage Data 75T060105-1015 No. 1
3/16 Laminate 4.0 Dia Damage Hole

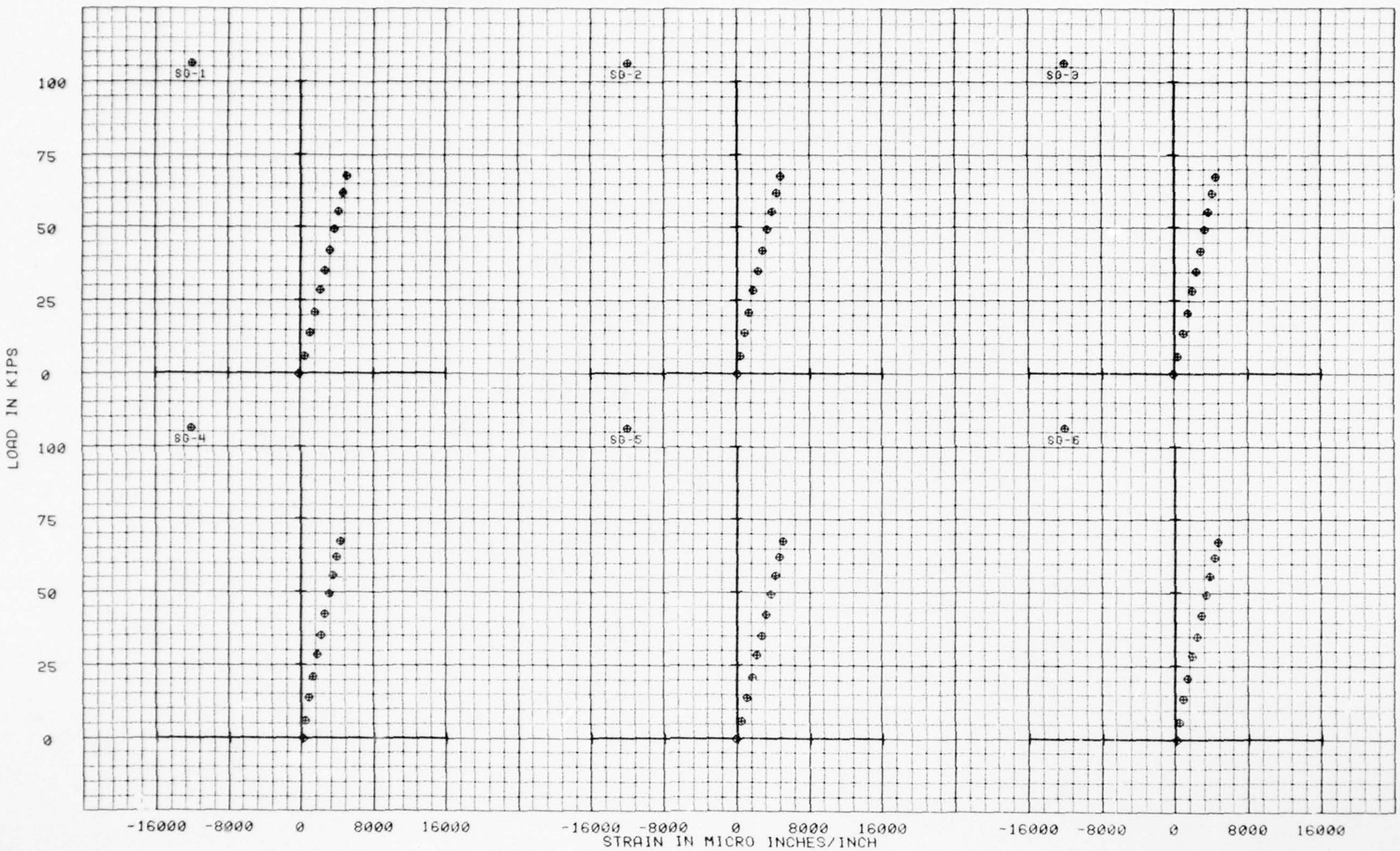


FIGURE B-12 Strain Gage Data 75T060105-1015 No. 2
3/16 Laminate 4.0 Dia Damage Hole

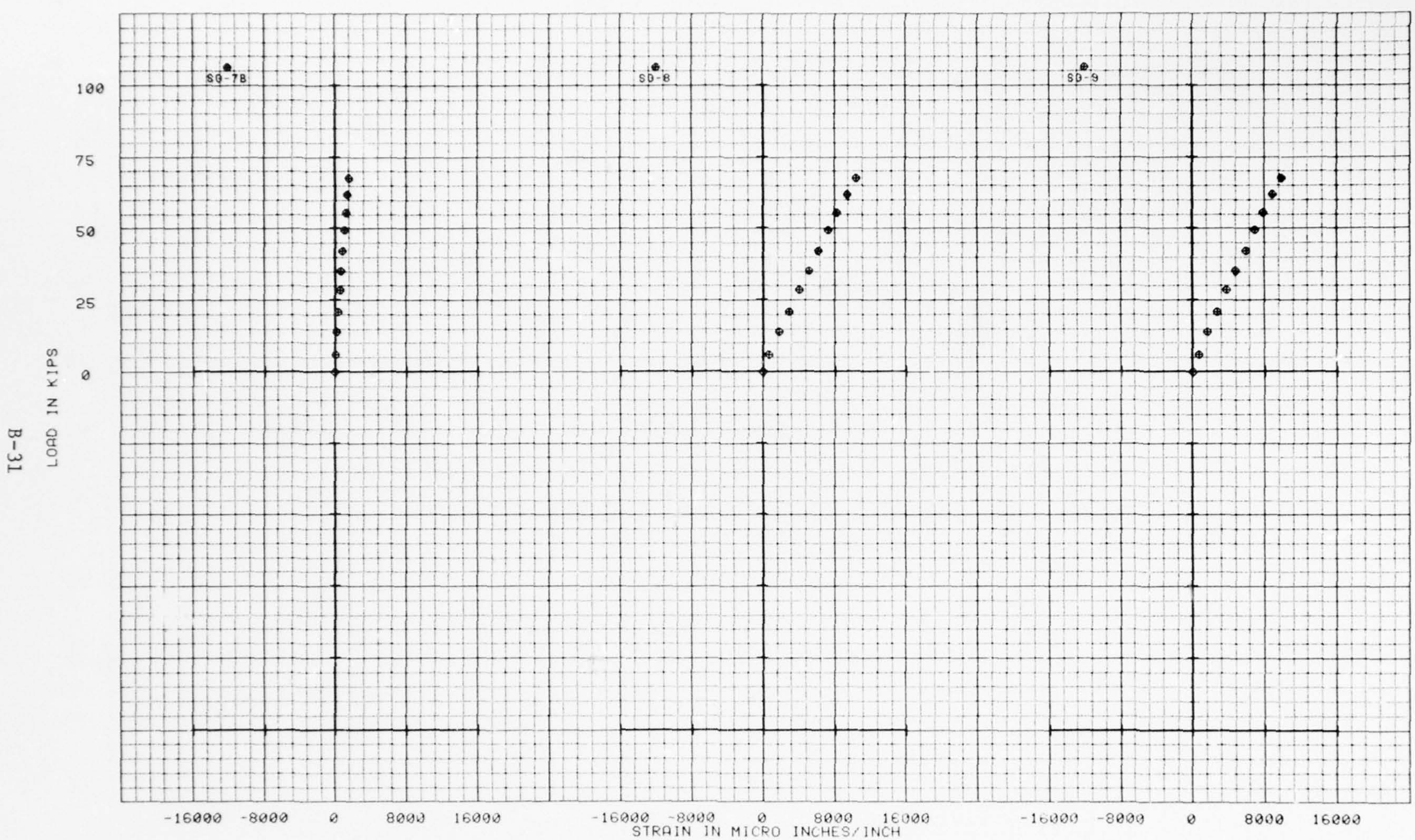


FIGURE B-12 Continued Strain Gage Data 75T060105-1015 No. 2
3/16 Laminate 4.0 Dia Damage Hole

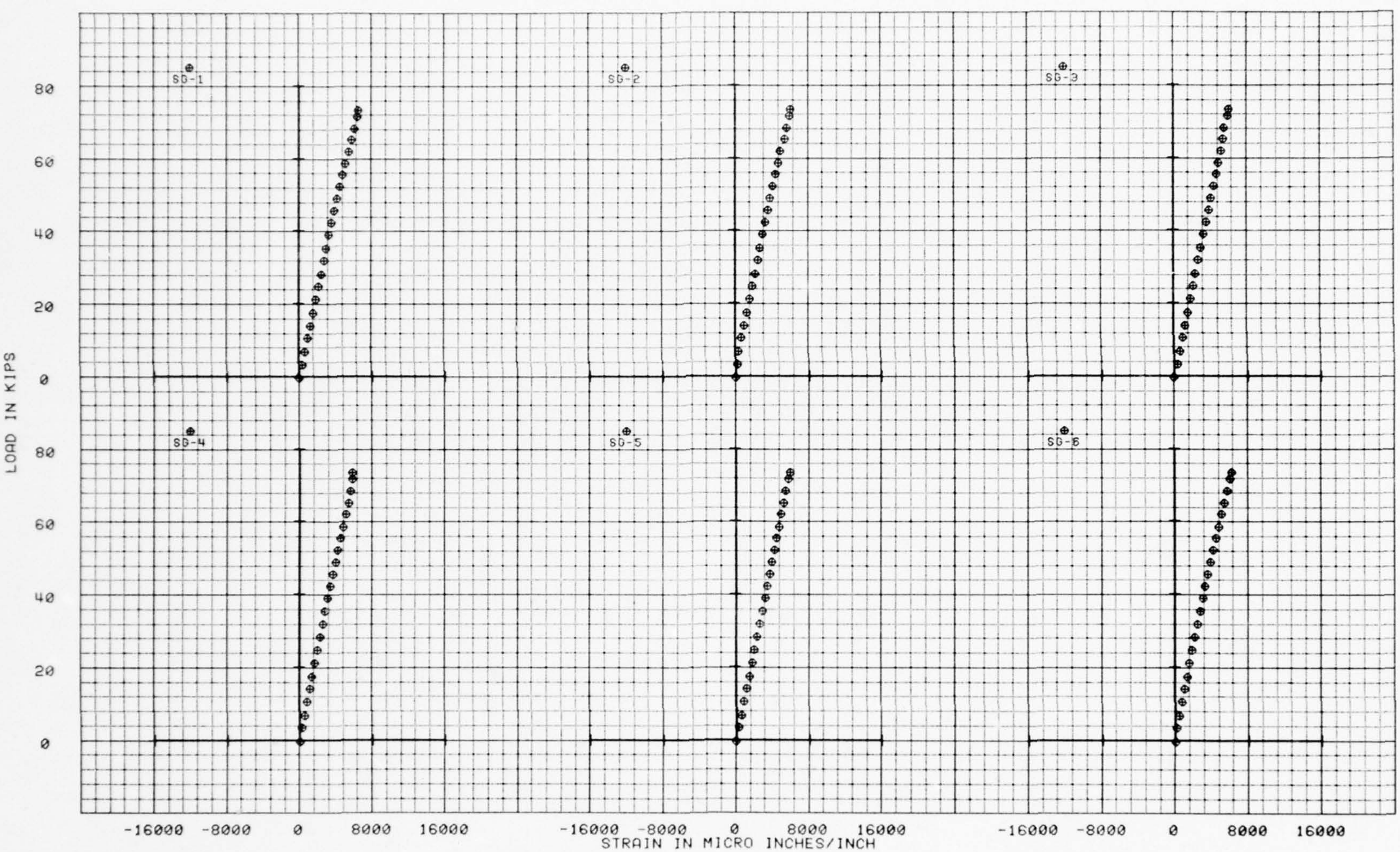


FIGURE B-13 Strain Gage Data 75T060105-1017
3/16 Laminate OFF Axis

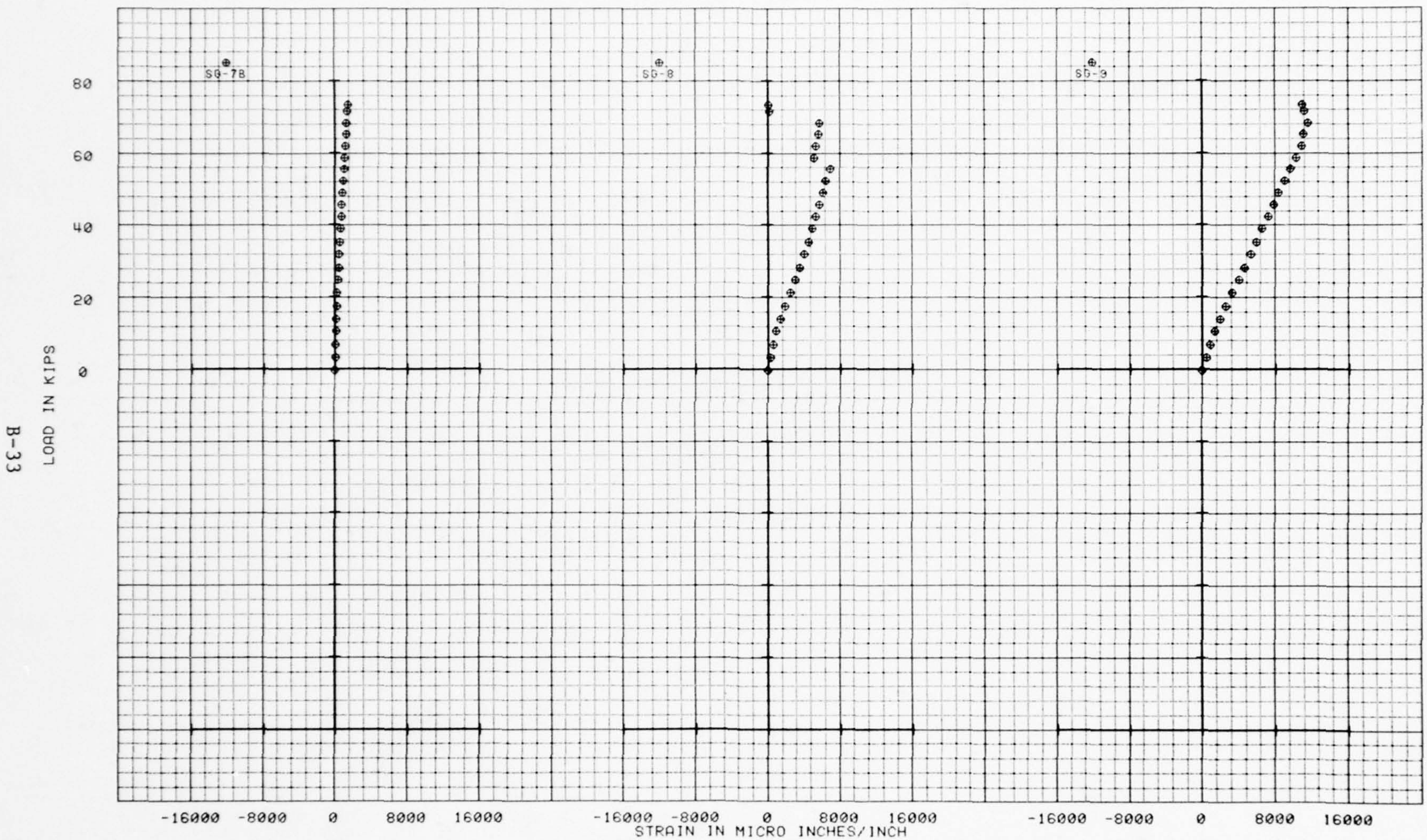


FIGURE B-13 Continued Strain Gage Data 75T060105-1017
3/16 Laminate OFF Axis

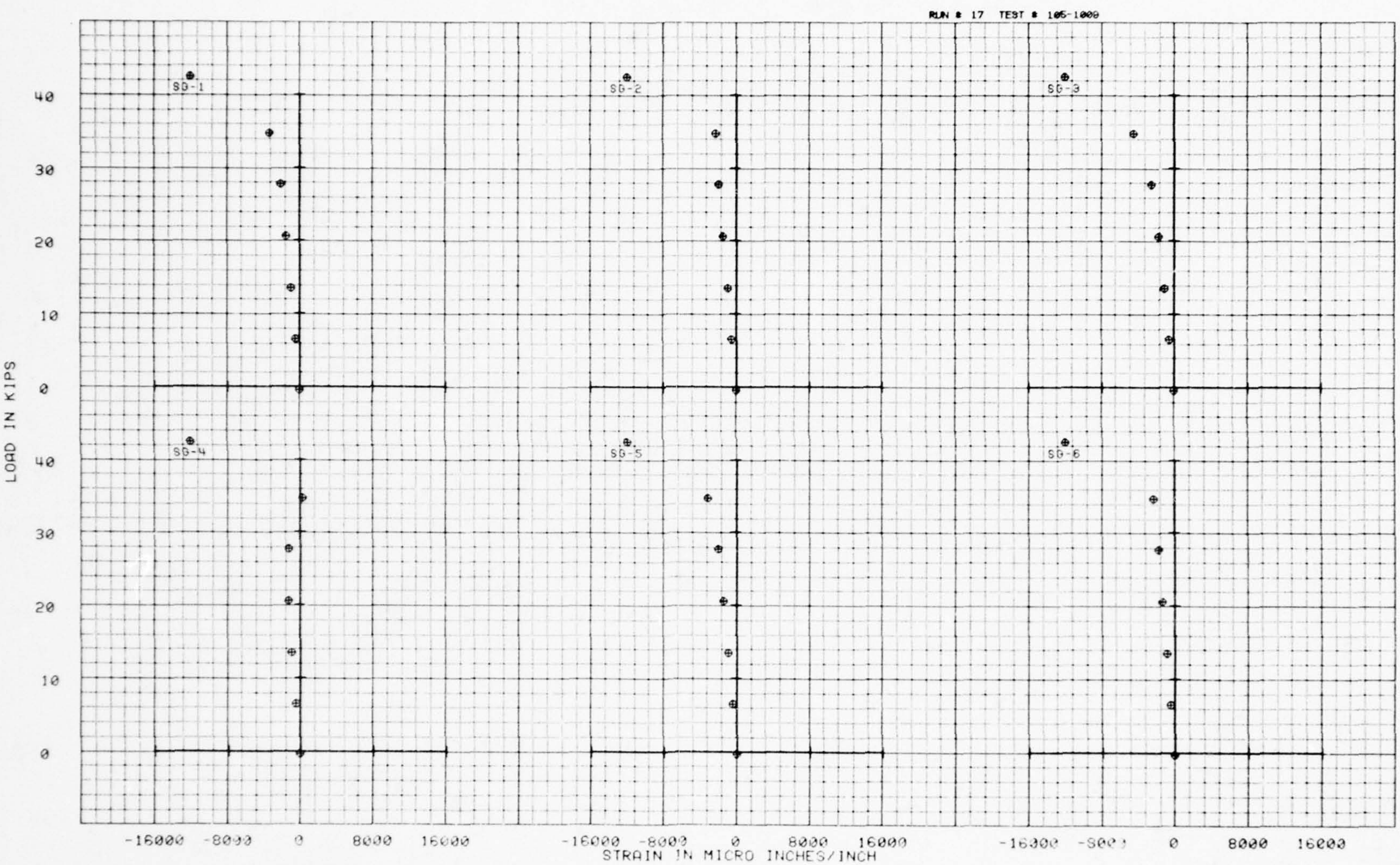


FIGURE B-14 Strain Gage Data 75T060105-1009
3/16 Laminate Compression

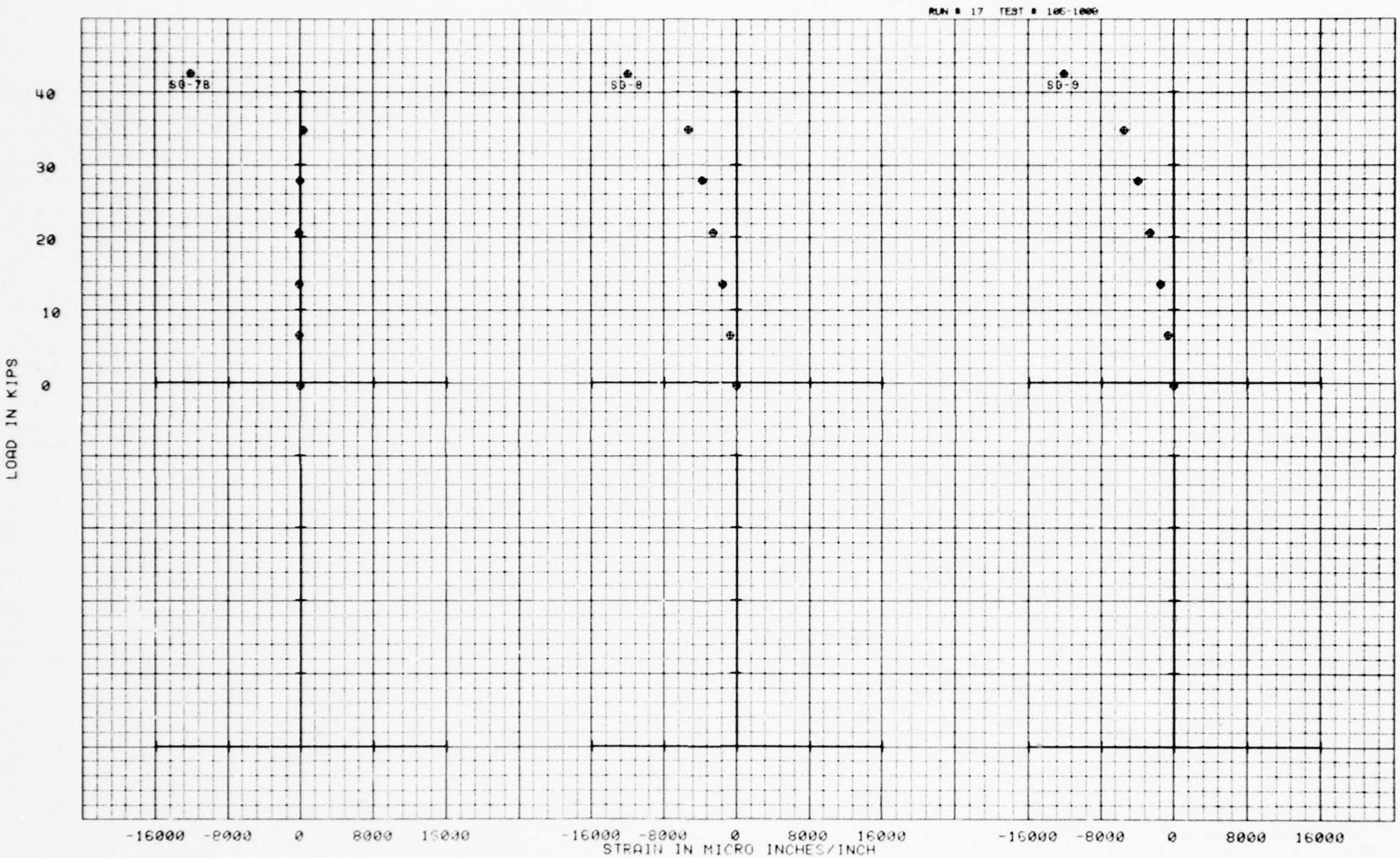


FIGURE B-14 Continued Strain Gage Data 75T060105-1009
3/16 Laminate Compression

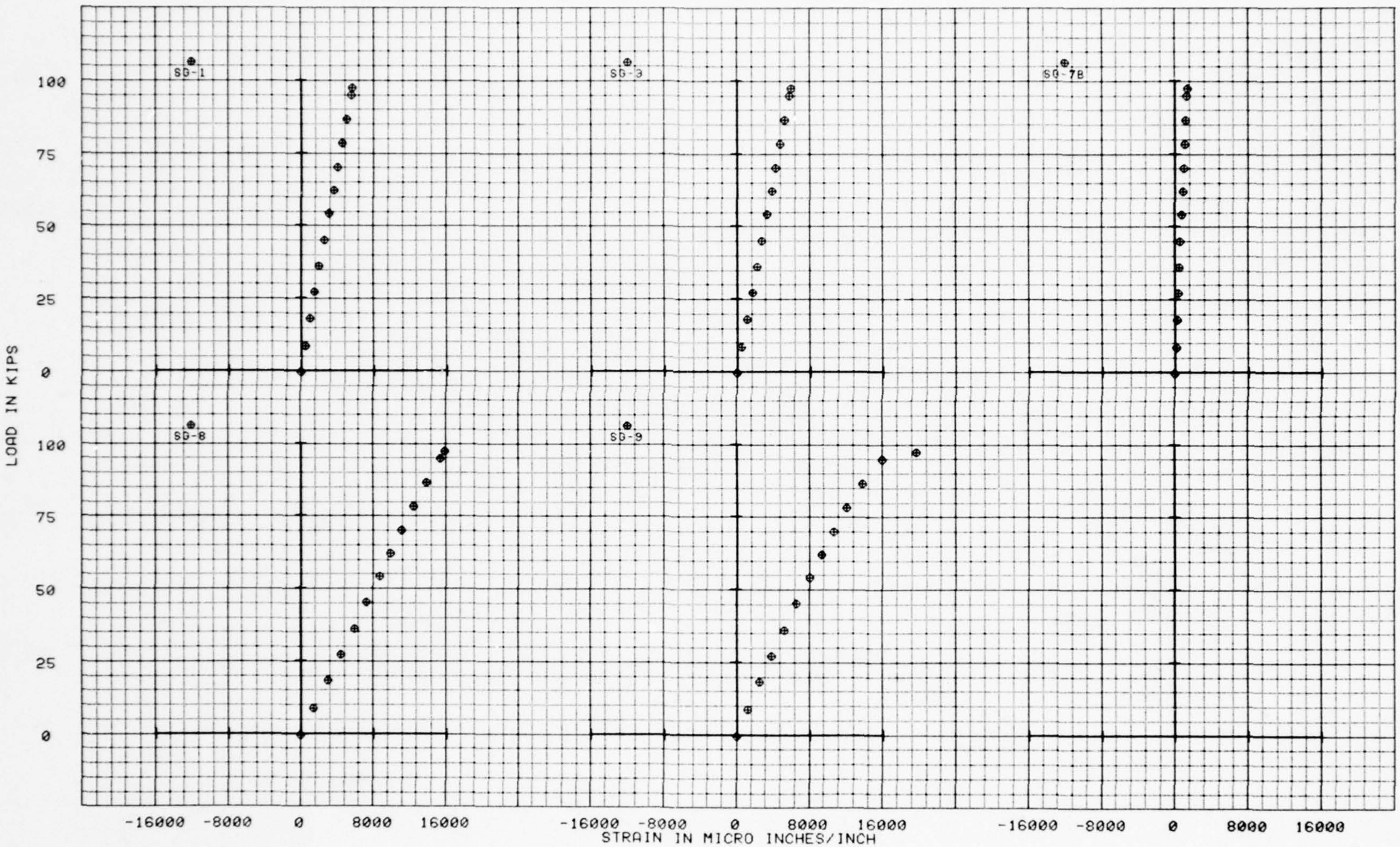


FIGURE B-15 Strain Gage Data 75T060106-1019
1/2 Laminate 1.0 Dia Damage Hole

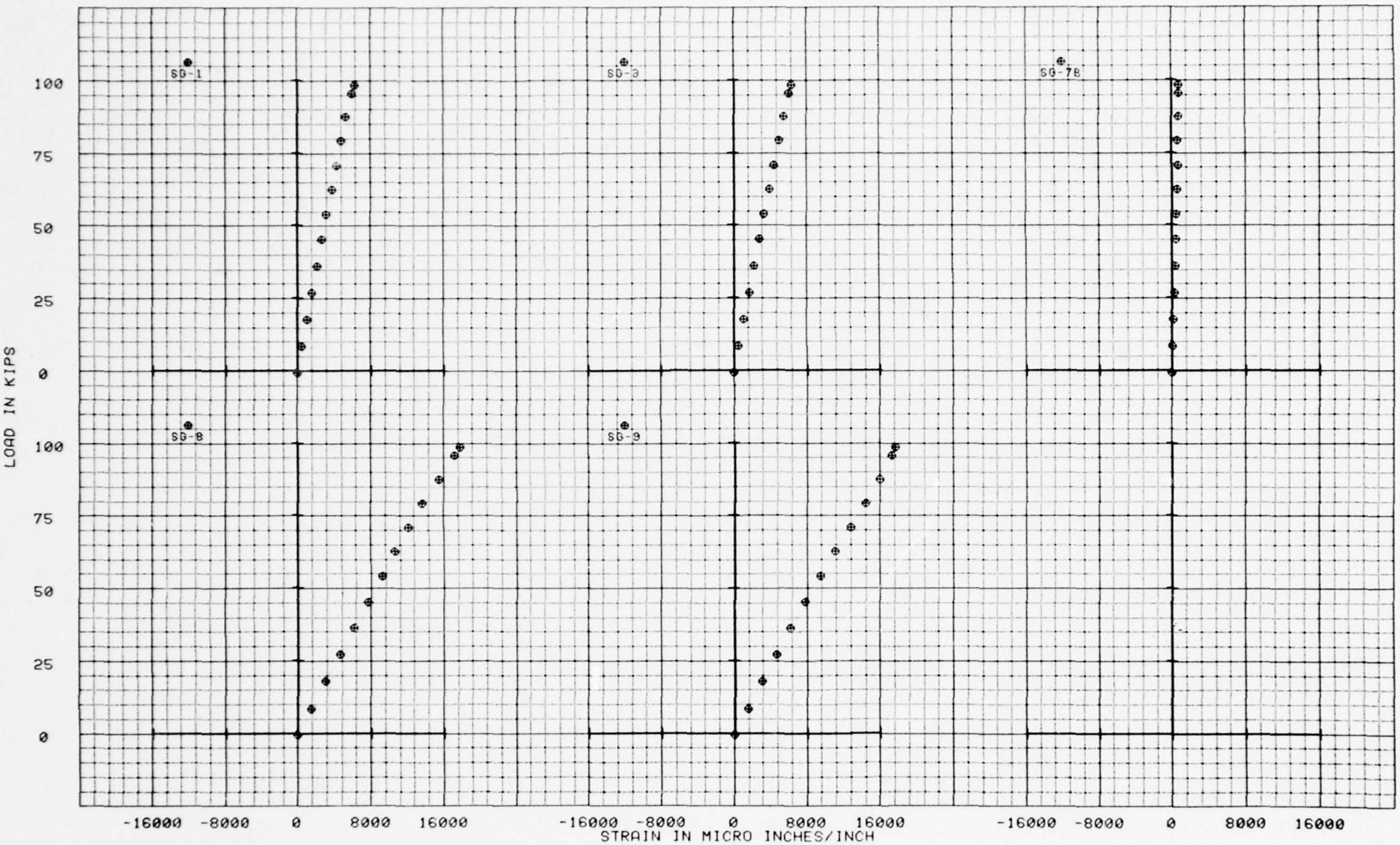


FIGURE B-16 Strain Gage Data 75T060106-1021 No. 1
1/2 Laminate 1.0 Dia Damage Hole

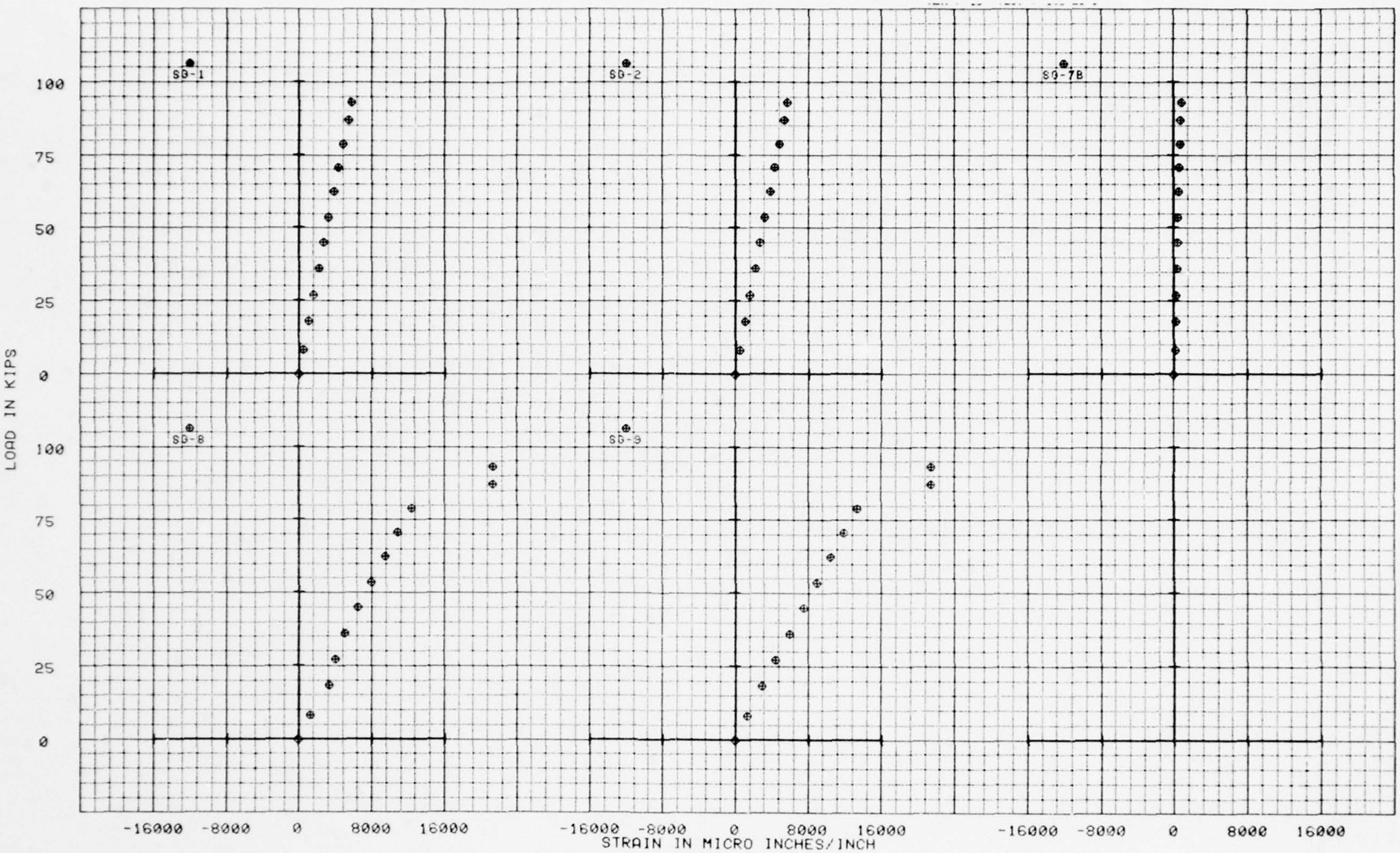


FIGURE B-17 Strain Gage Data 75T060106-1021 No. 2
1/2 Laminate 1.0 Dia Damage Hole

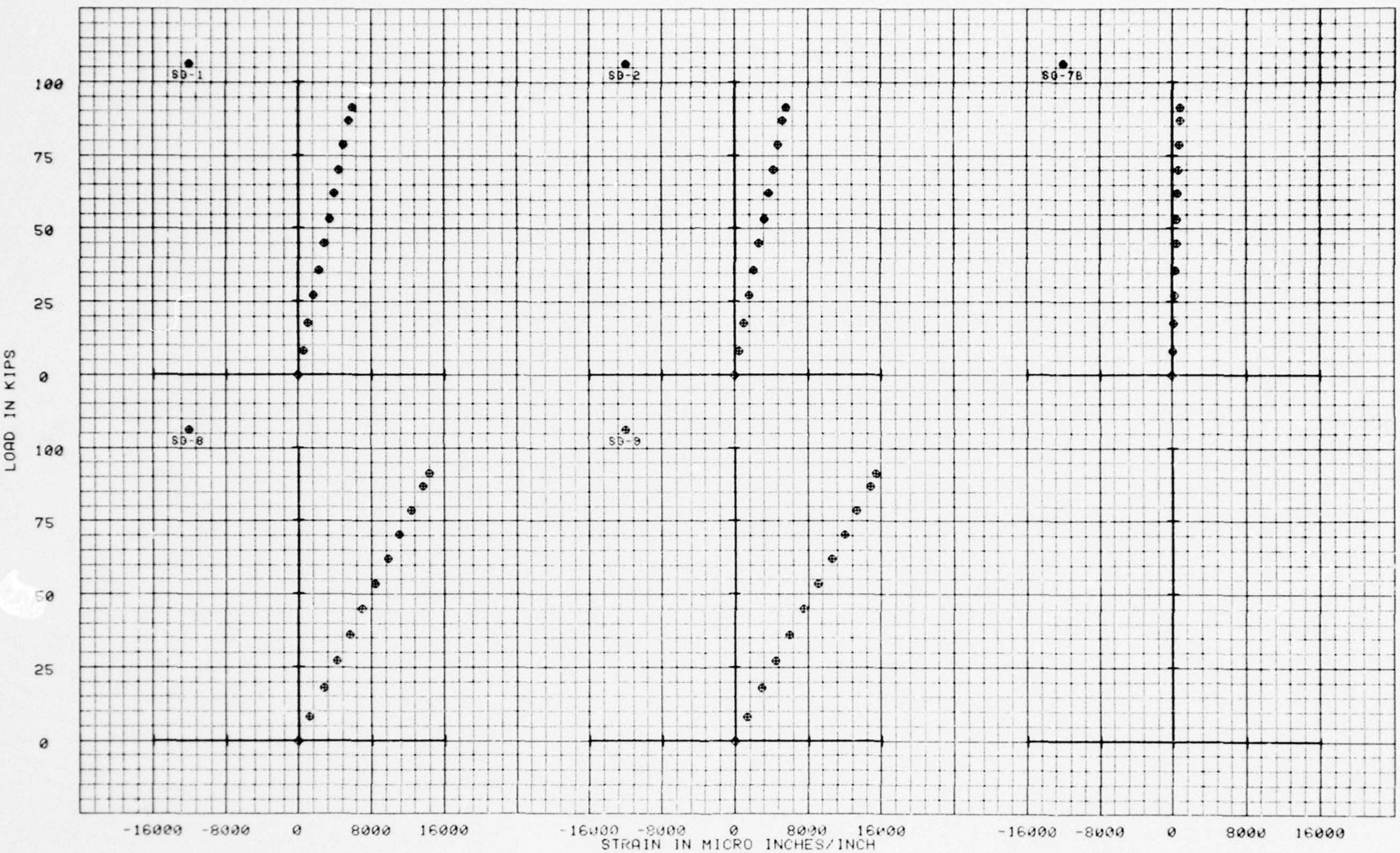


FIGURE B-18 Strain Gage Data 75T060106-1021 No. 3
1/2 Laminate 1.0 Dia Damage Hole

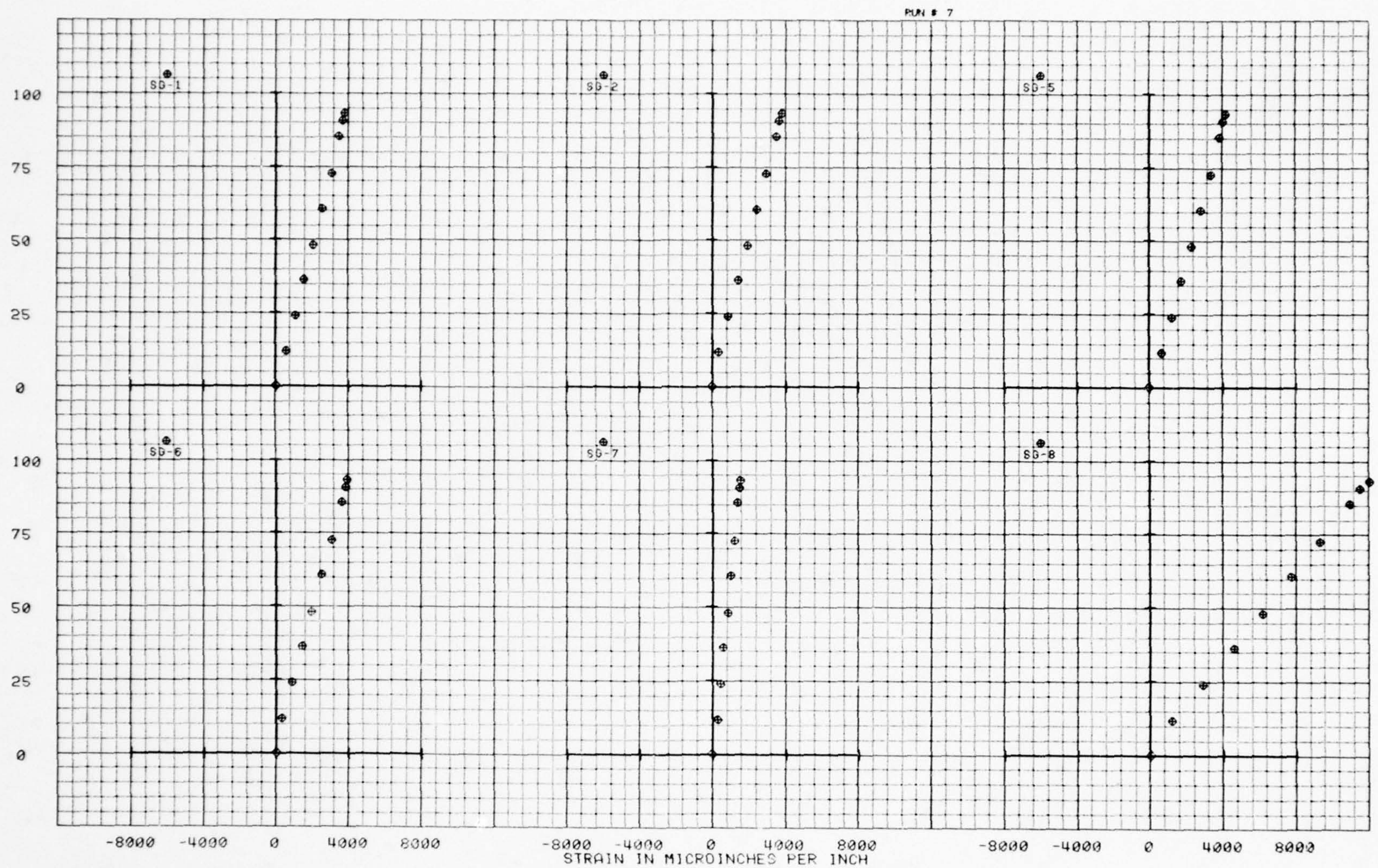


FIGURE B-19 Strain Gage Data 75T060106-1005
1/2 Laminate 2.5 Dia Damage Hole

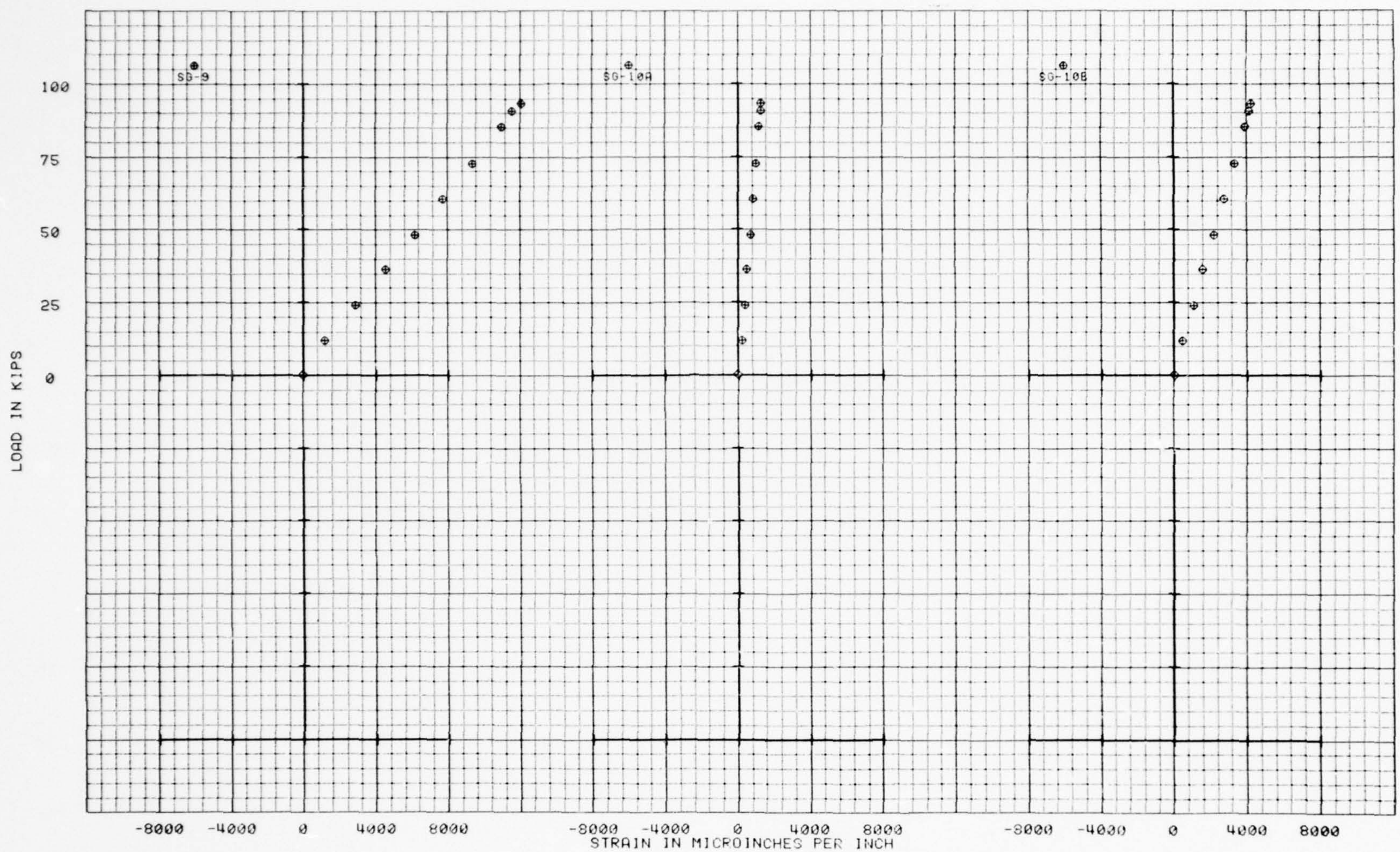


FIGURE B-19 Continued Strain Gage Data 75T060106-1005
1/2 Laminate 2.5 Dia Damage Hole

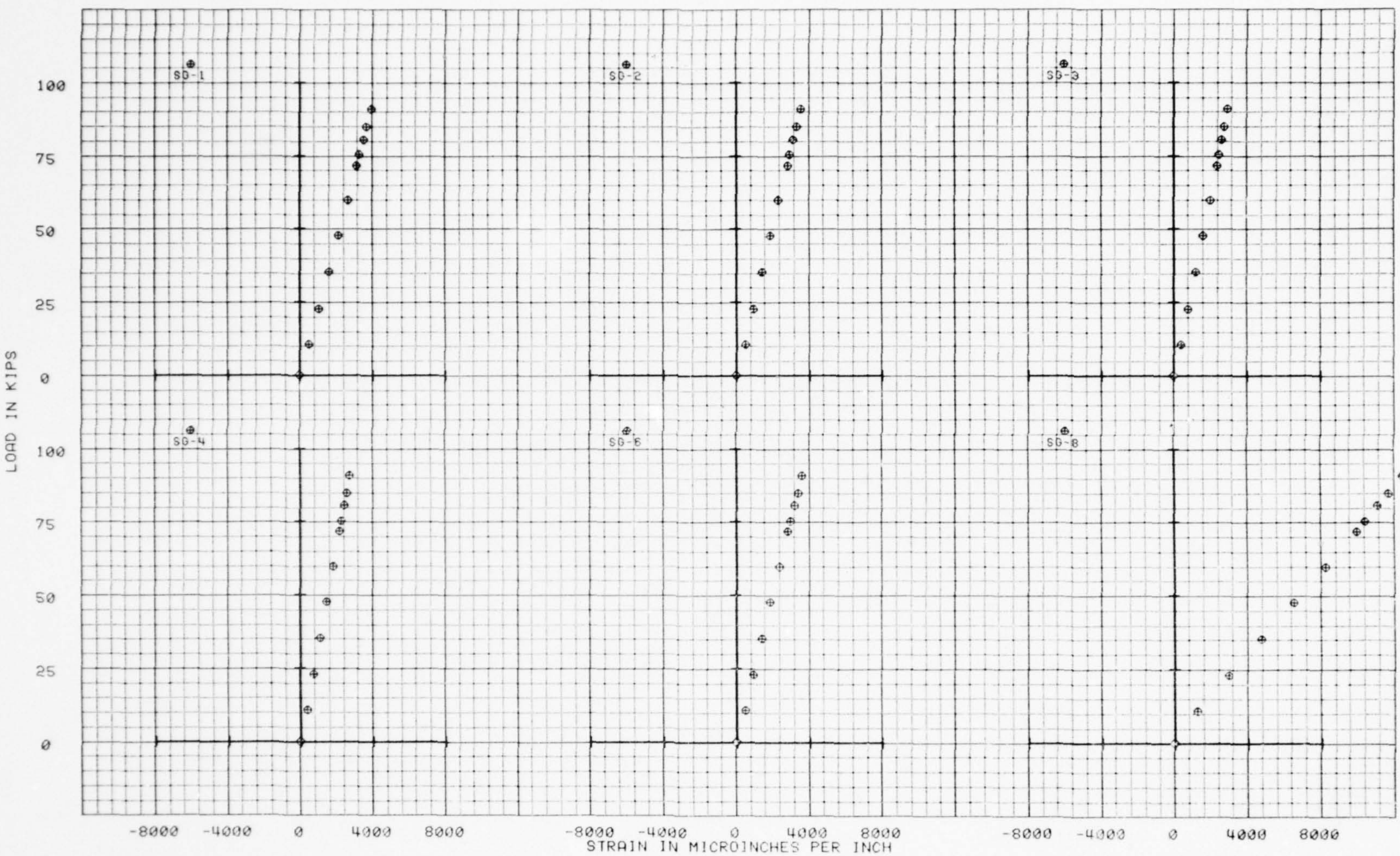


FIGURE B-20 Strain Gage Data 75T060106-1007 No. 1
1/2 Laminate 2.5 Dia Damage Hole

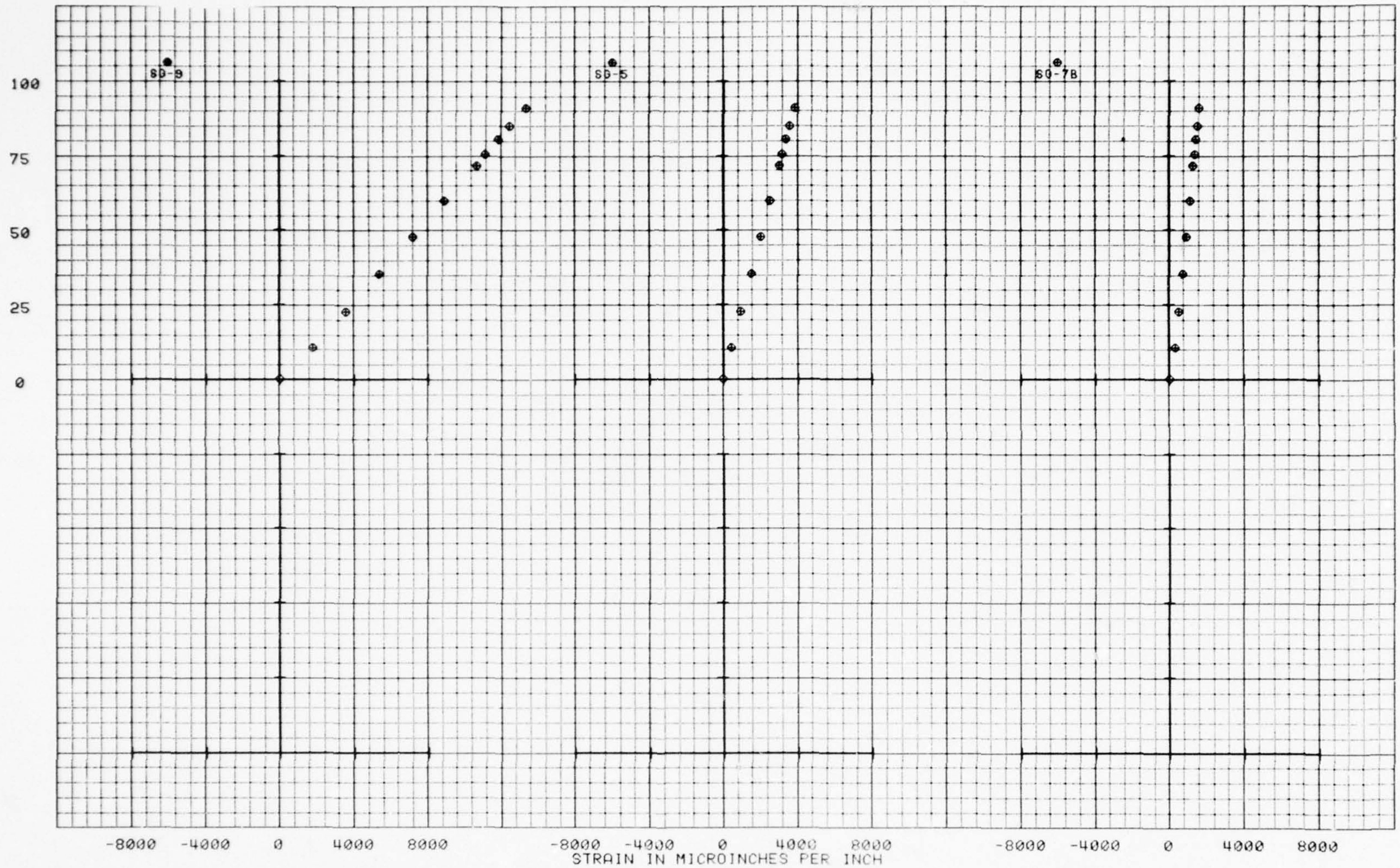


FIGURE B-20 Continued Strain Gage Data 75T060106-1007 No. 1
1/2 Laminate 2.5 Dia Damage Hole

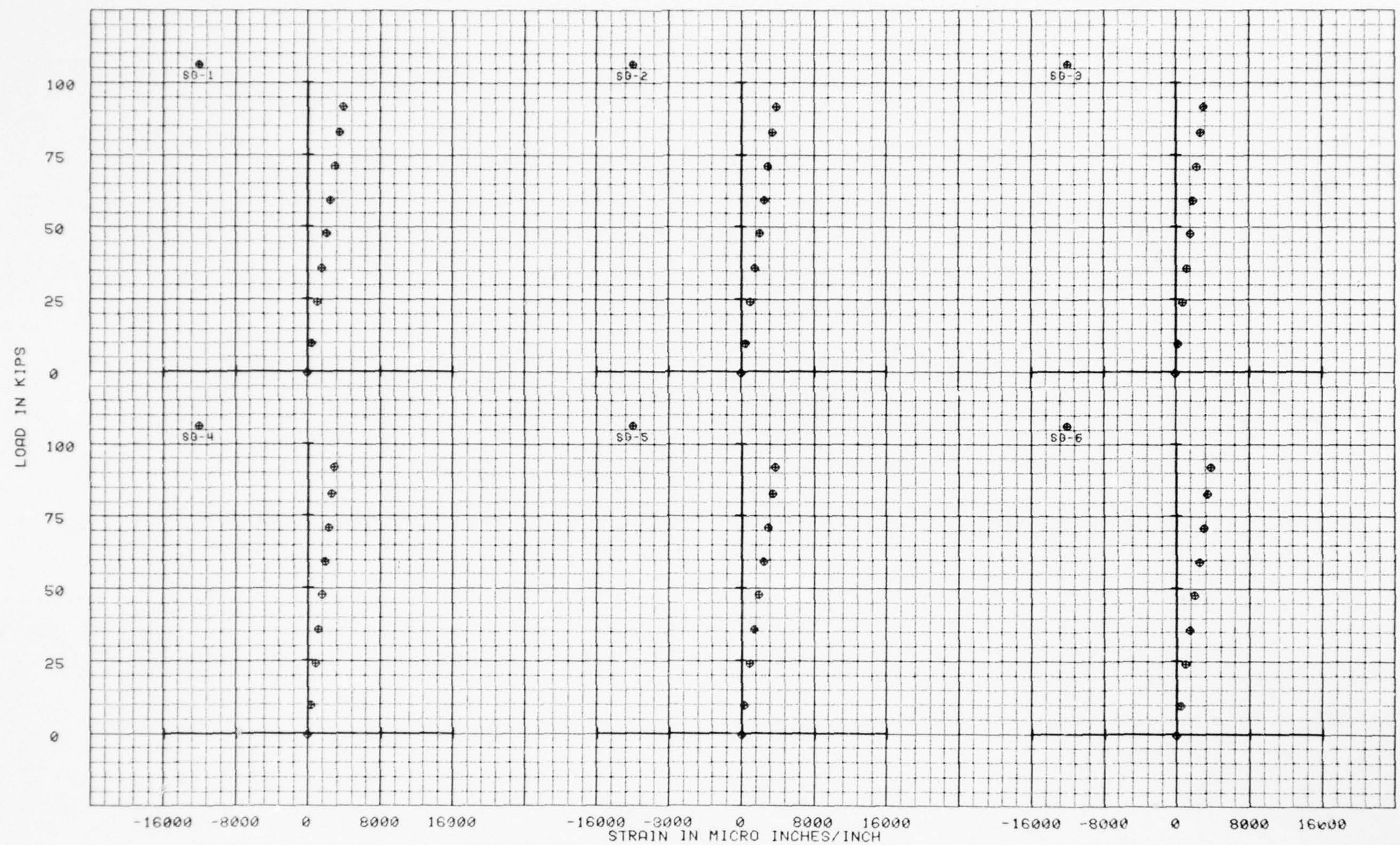


FIGURE B-21 Strain Gage Data 75T060106-1007 No. 2
1/2 Laminate 2.5 Dia Damage Hole

AD-A067 923 MCDONNELL AIRCRAFT CO ST LOUIS MO F/G 11/2

BOLTED FIELD REPAIR OF COMPOSITE STRUCTURES.(U)

MAR 79 J B WATSON, D A GLAESER, F L HARVEY N62269-77-C-0366

UNCLASSIFIED MDC-A5583 NADC-77109-30 NL

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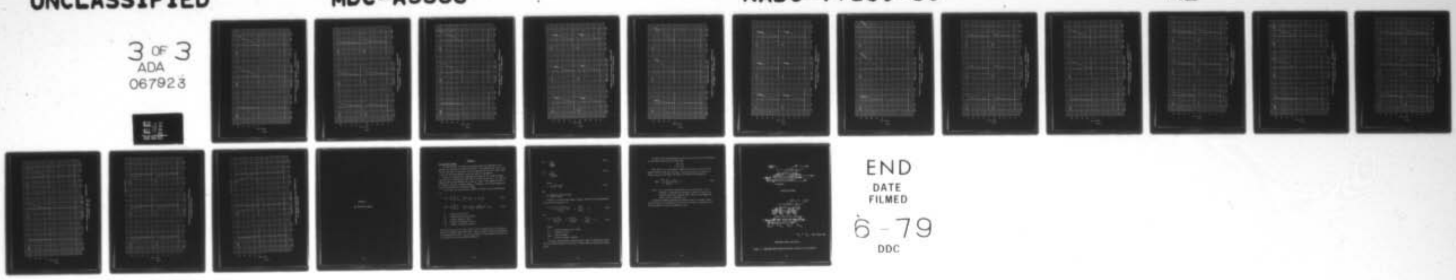

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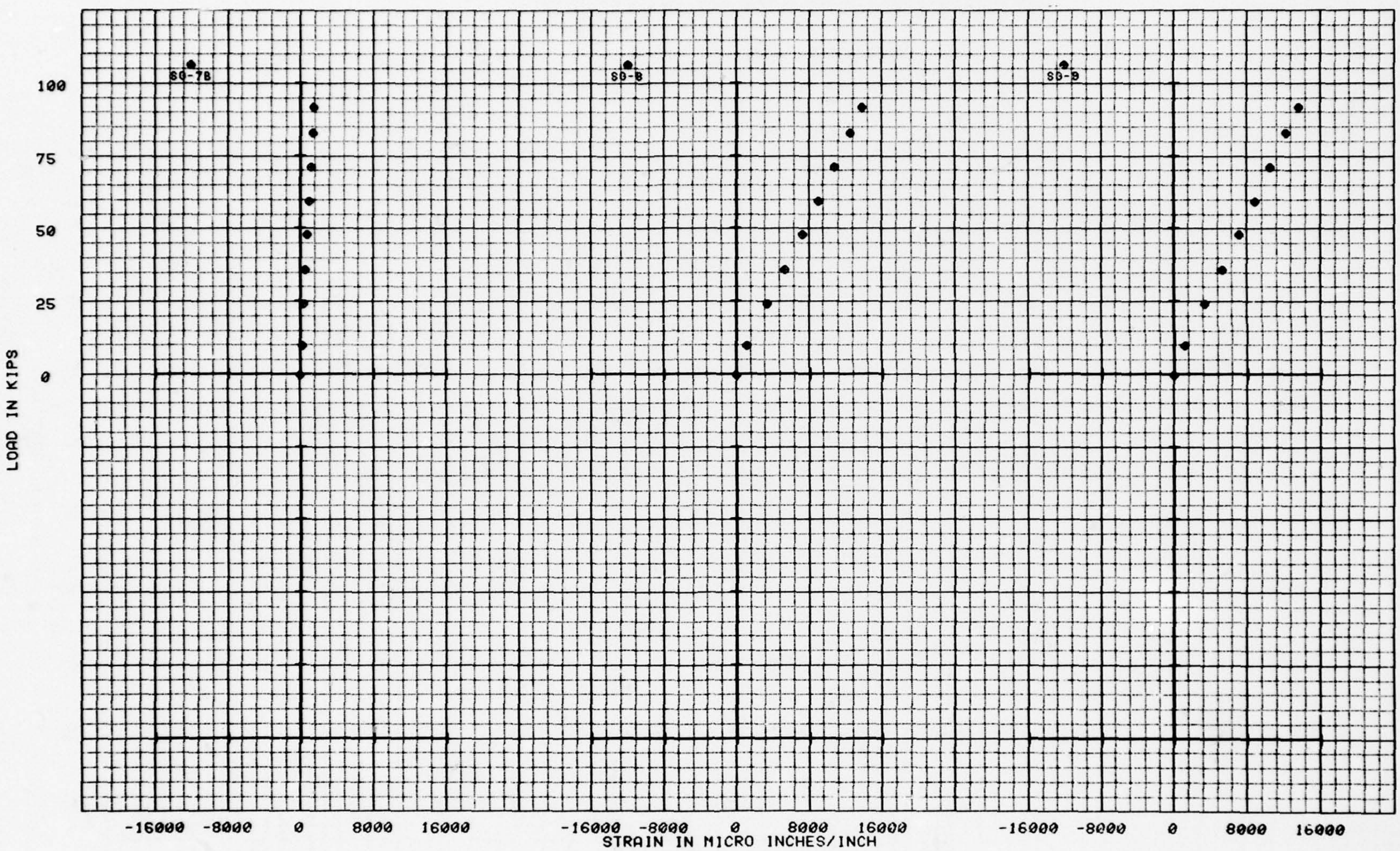


FIGURE B-21 Continued Strain Gage Data 75T060106-1007 No. 2
1/2 Laminate 2.5 Dia Damage Hole

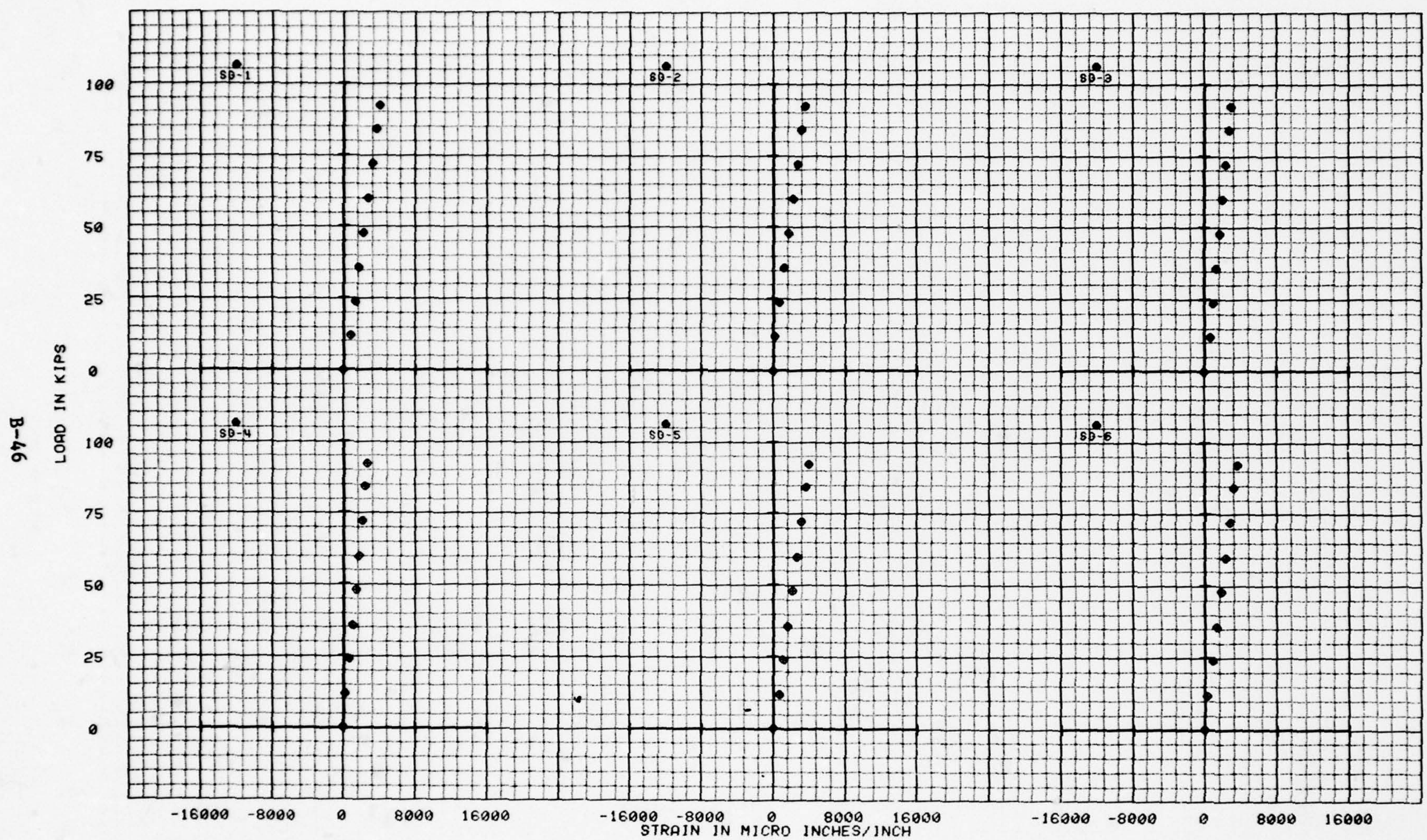


FIGURE B-22 Strain Gage Data 75T060106-1017
1/2 Laminate 2.5 Dia Damage Hole

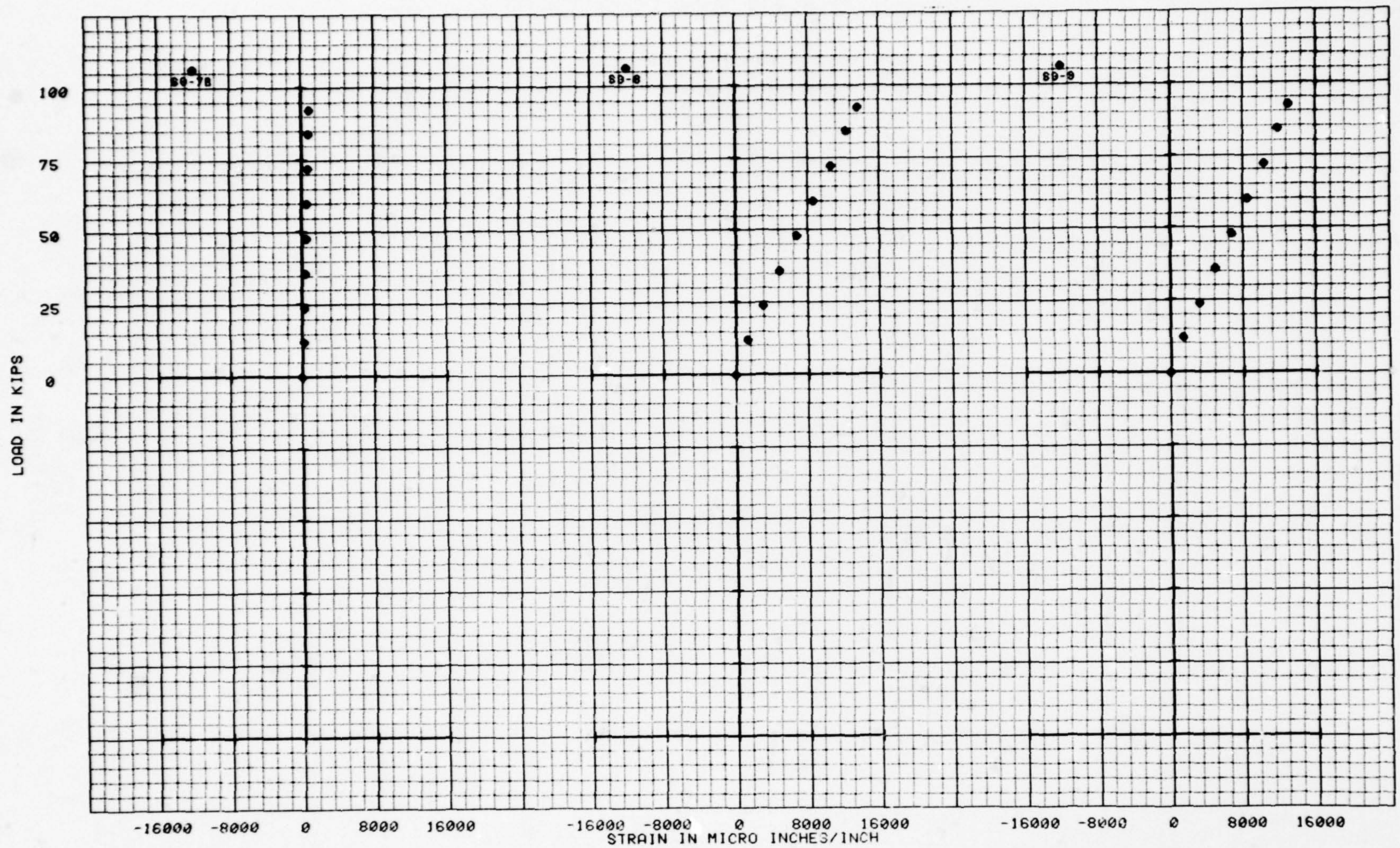


FIGURE B-22 Continued Strain Gage Data 75T060106-1017
1/2 Laminate 2.5 Dia Damage Hole

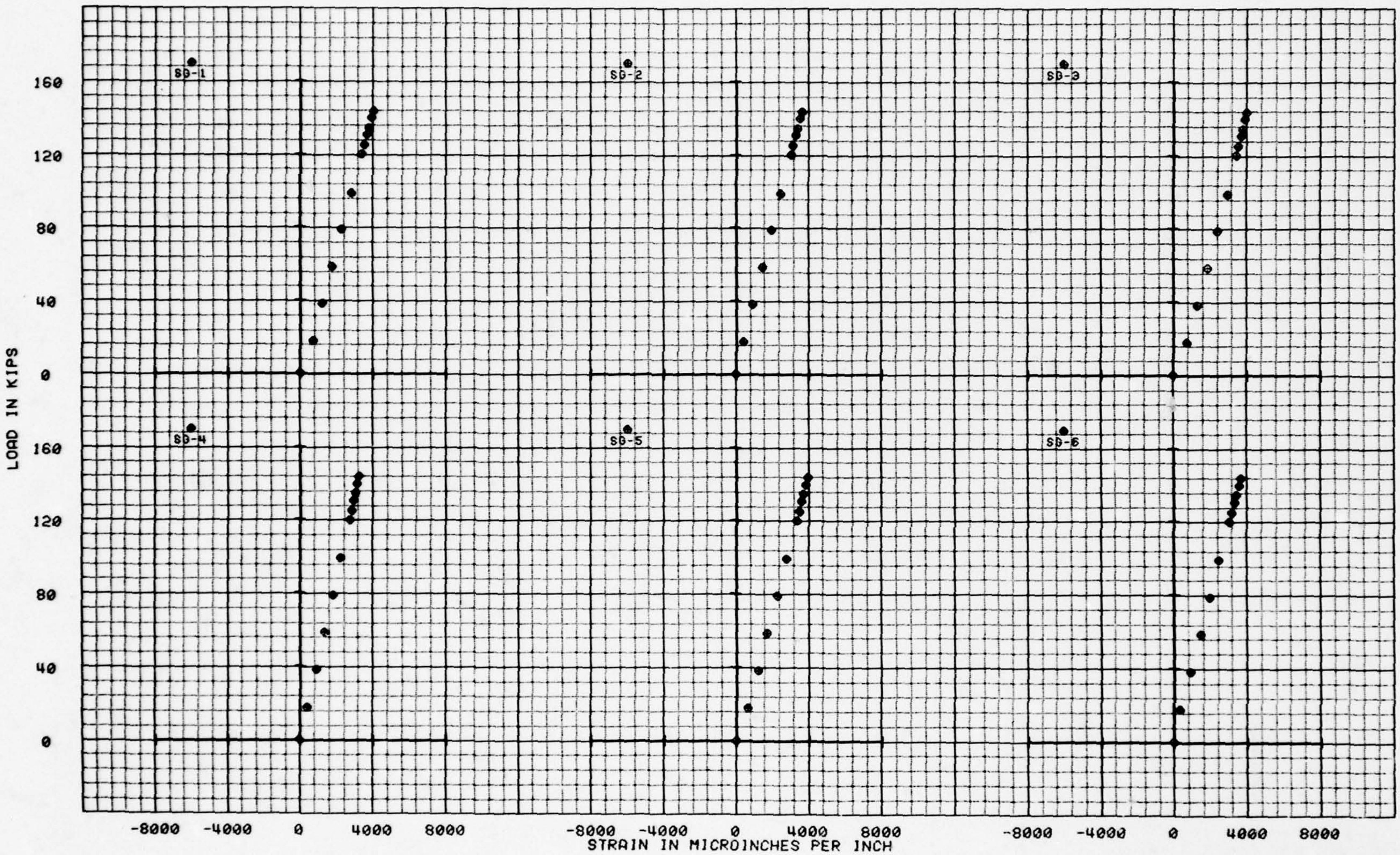


FIGURE B-23 Strain Gage Data 75T060106-1009
1/2 Laminate 4.0 Dia Damage Hole

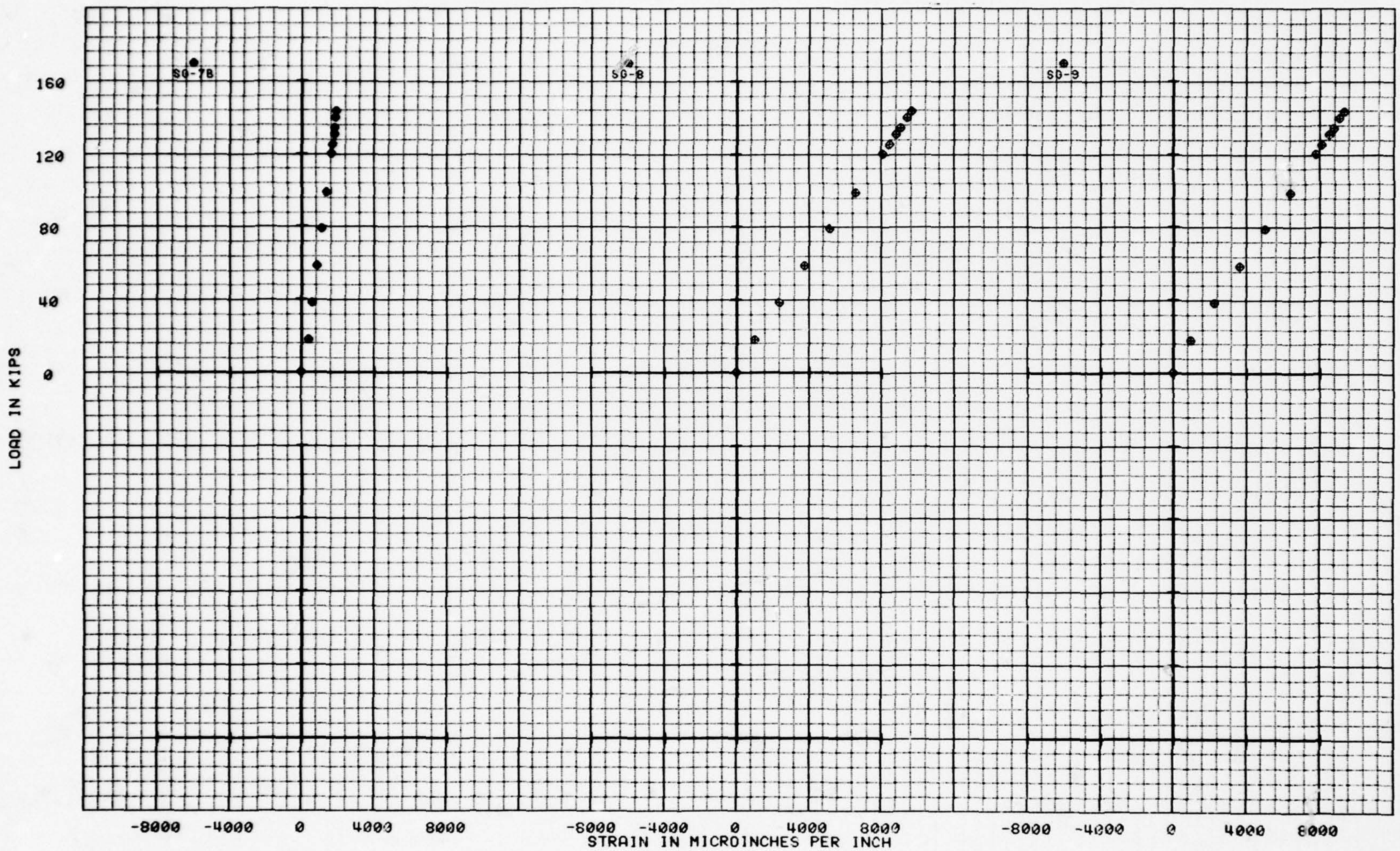


FIGURE B-23 Continued Strain Gage Data 75T060106-1009
1/2 Laminate 4.0 Dia Damage Hole

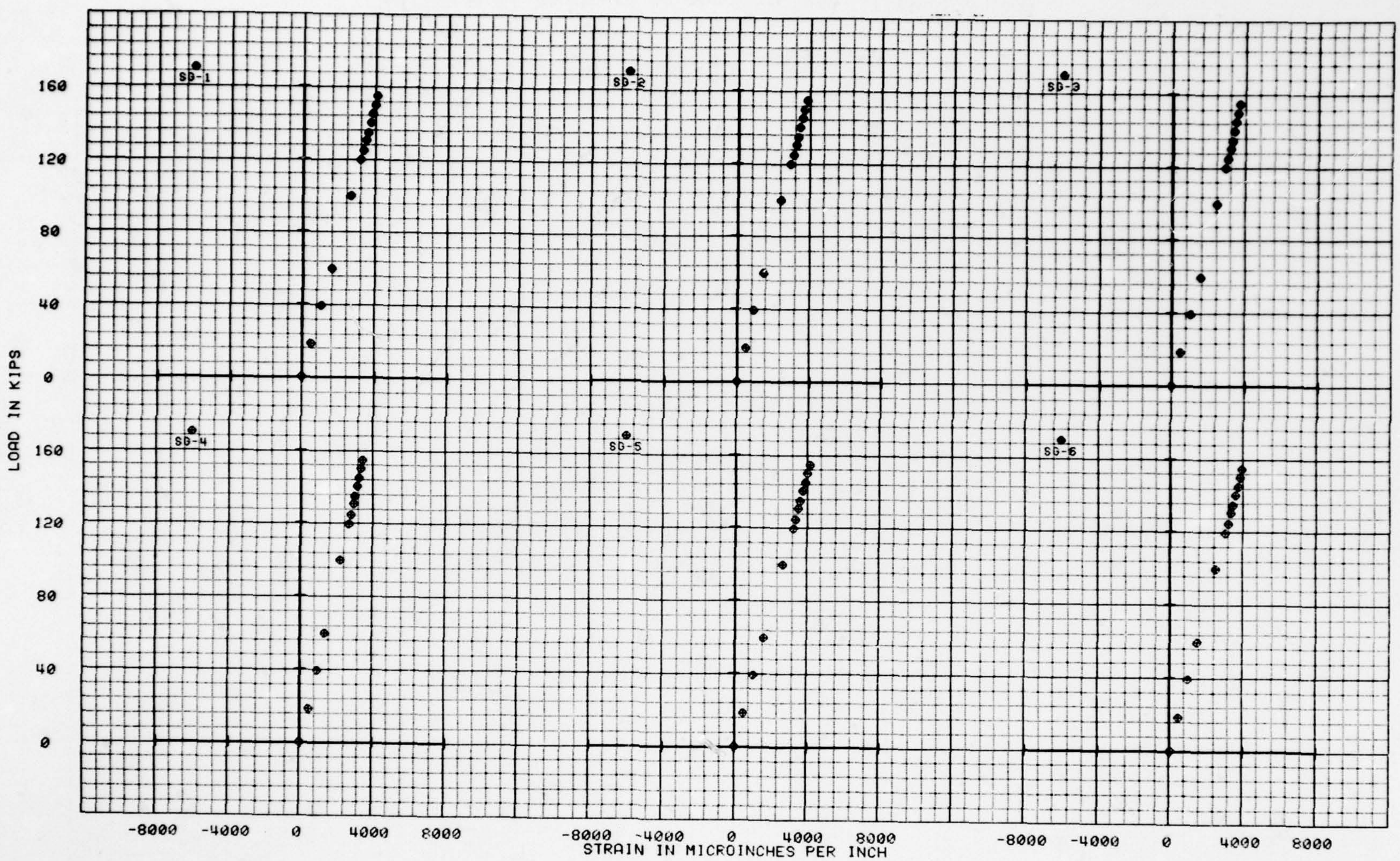


FIGURE B-24 Strain Gage Data 75T060106-1011 No. 1
1/2 Laminate 4.0 Dia Damage Hole

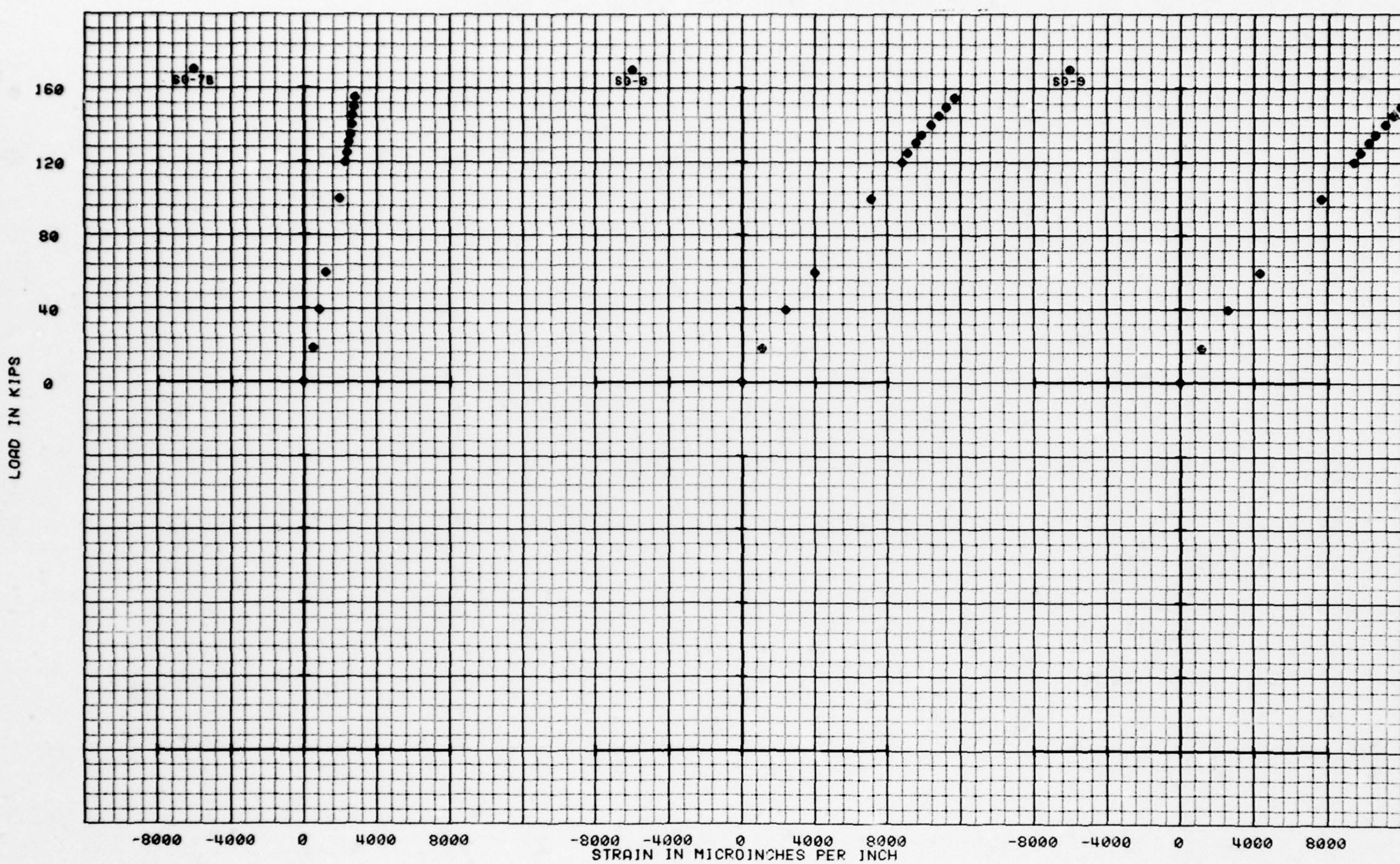


FIGURE B-24 Continued Strain Gage Data 75T060106-1011 No. 1
1/2 Laminate 4.0 Dia Damage Hole

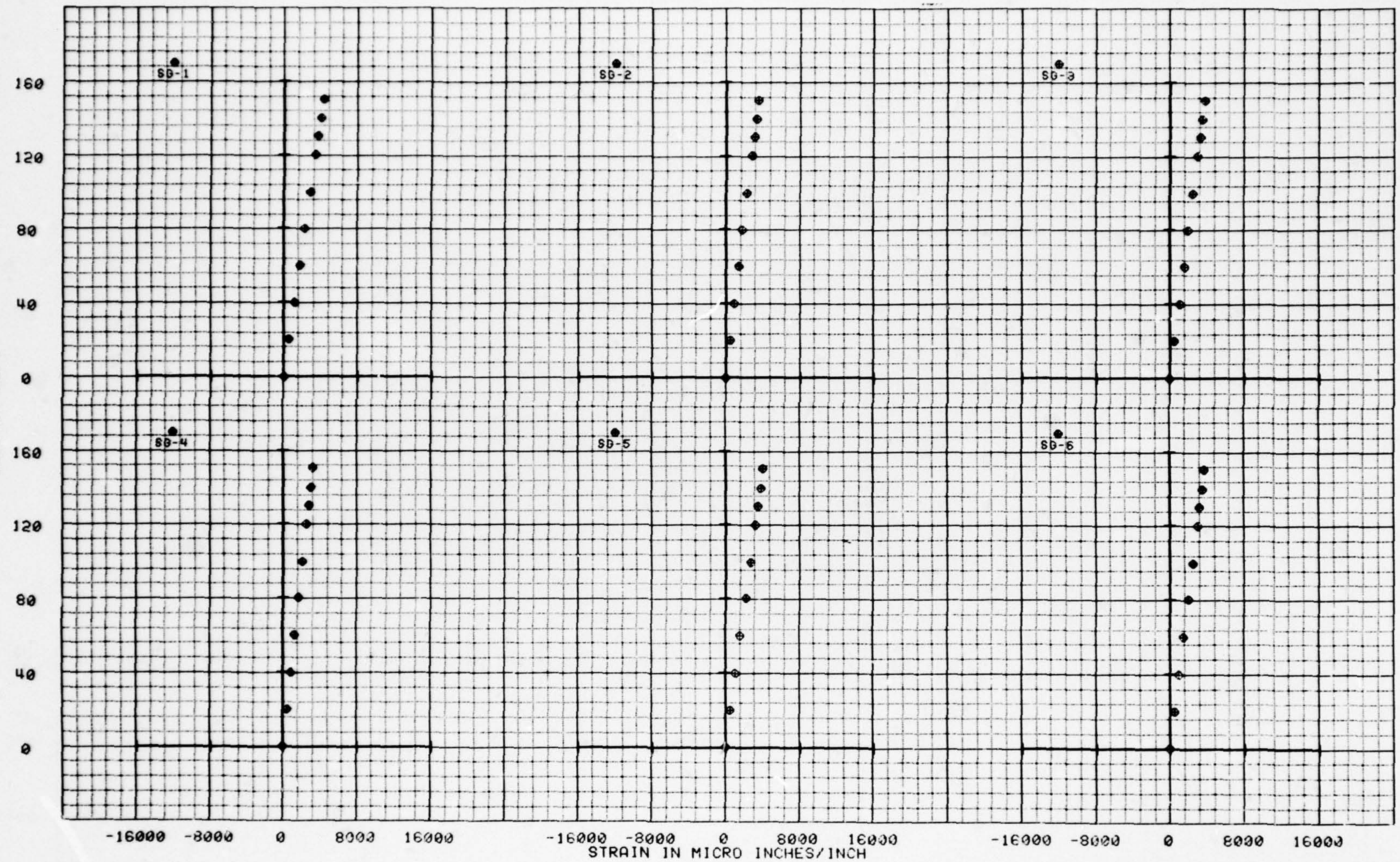


FIGURE B-25 Strain Gage Data 75T060106-1011 No. 2
1/2 Laminate 4.0 Dia Damage Hole

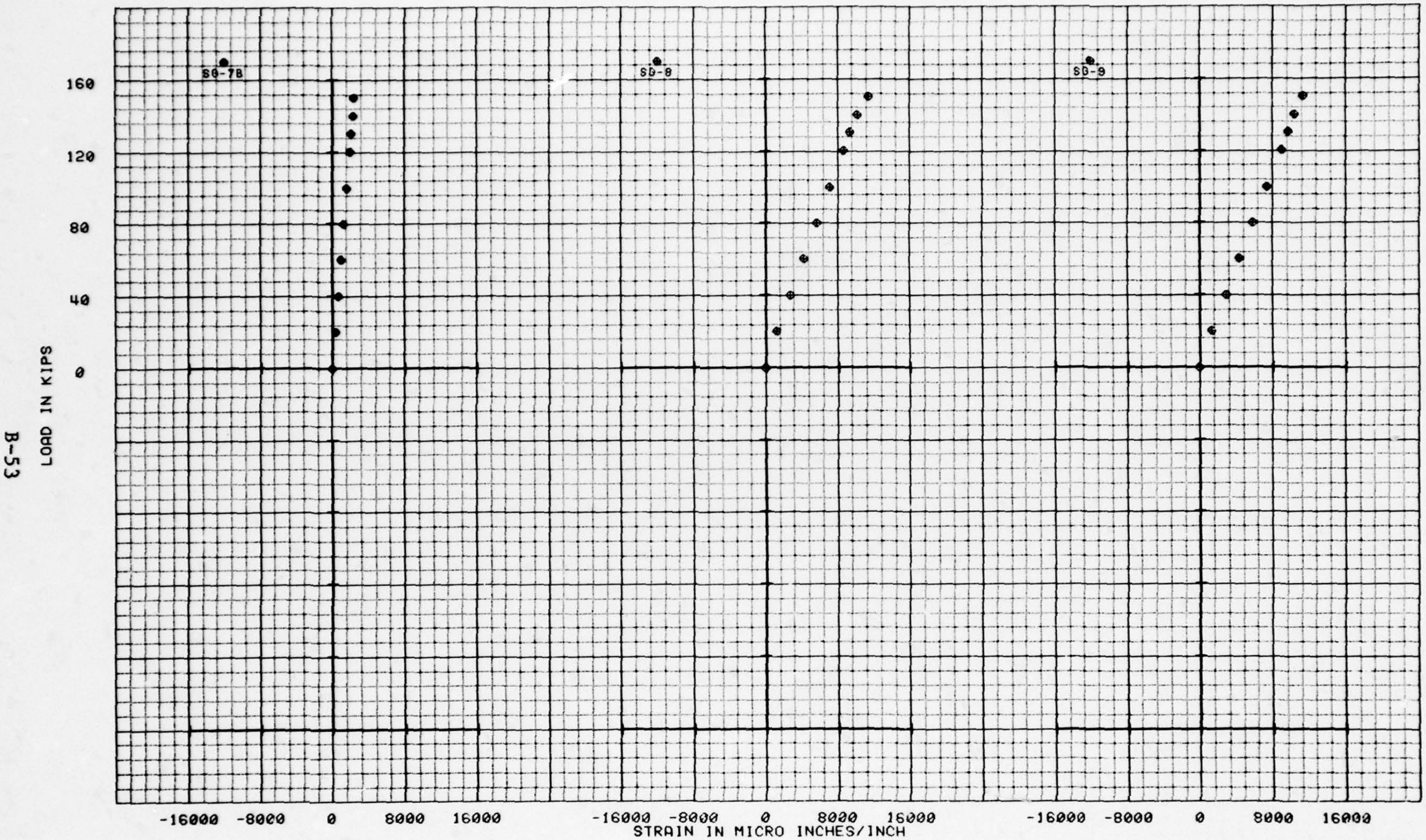


FIGURE B-25 Continued Strain Gage Data 75T060106-1011 No. 2
1/2 Laminate 4.0 Dia Damage Hole

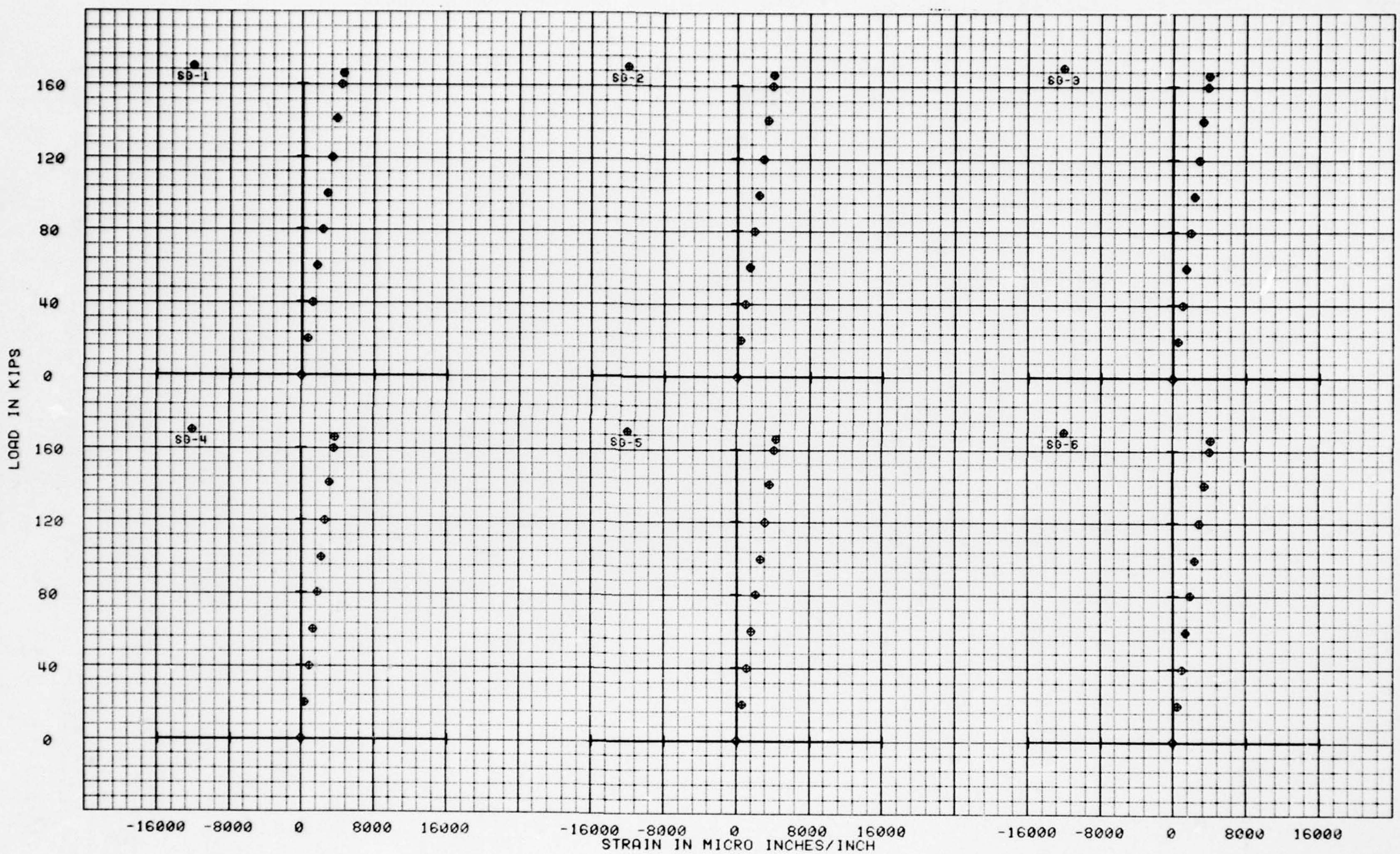


FIGURE B-26 Strain Gage Data 75T060106-1011 No. 3
1/2 Laminate 4.0 Dia Damage Hole

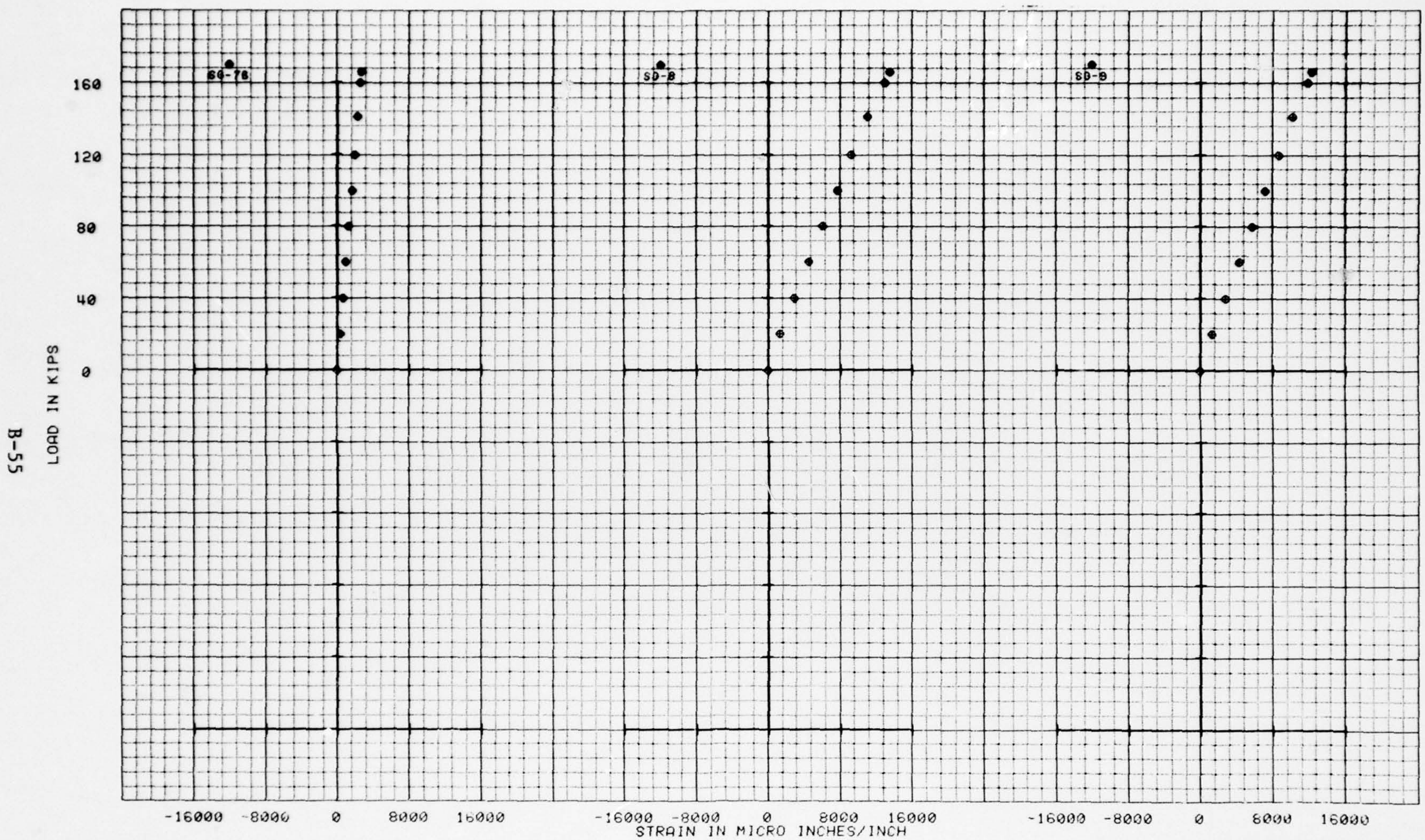


FIGURE B-26 Continued Strain Gage Data 75T060106-1011 No. 3
1/2 Laminate 4.0 Dia Damage Hole

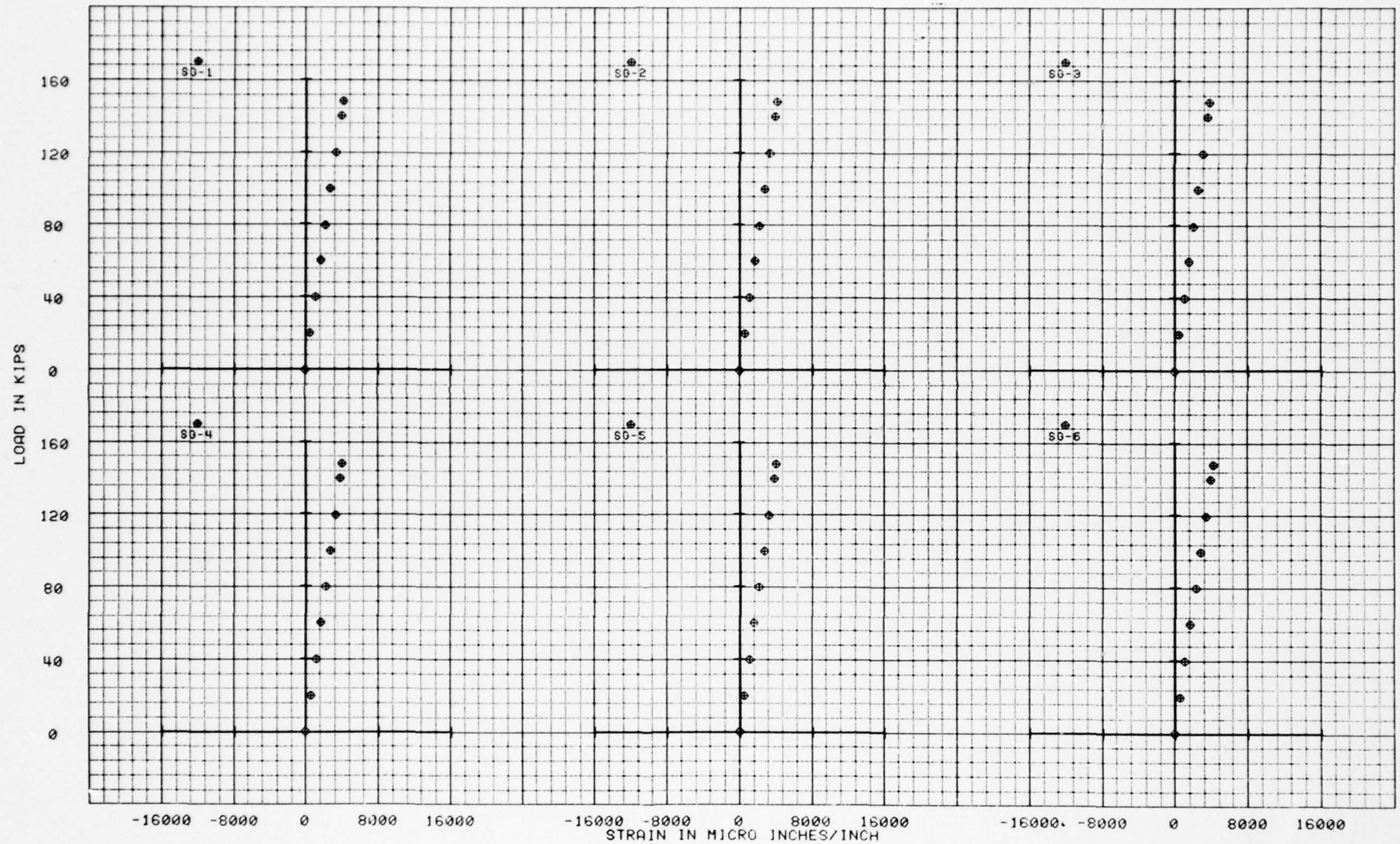


FIGURE B-27 Strain Gage Data 75T060106-1013
1/2 Laminate Off Axis

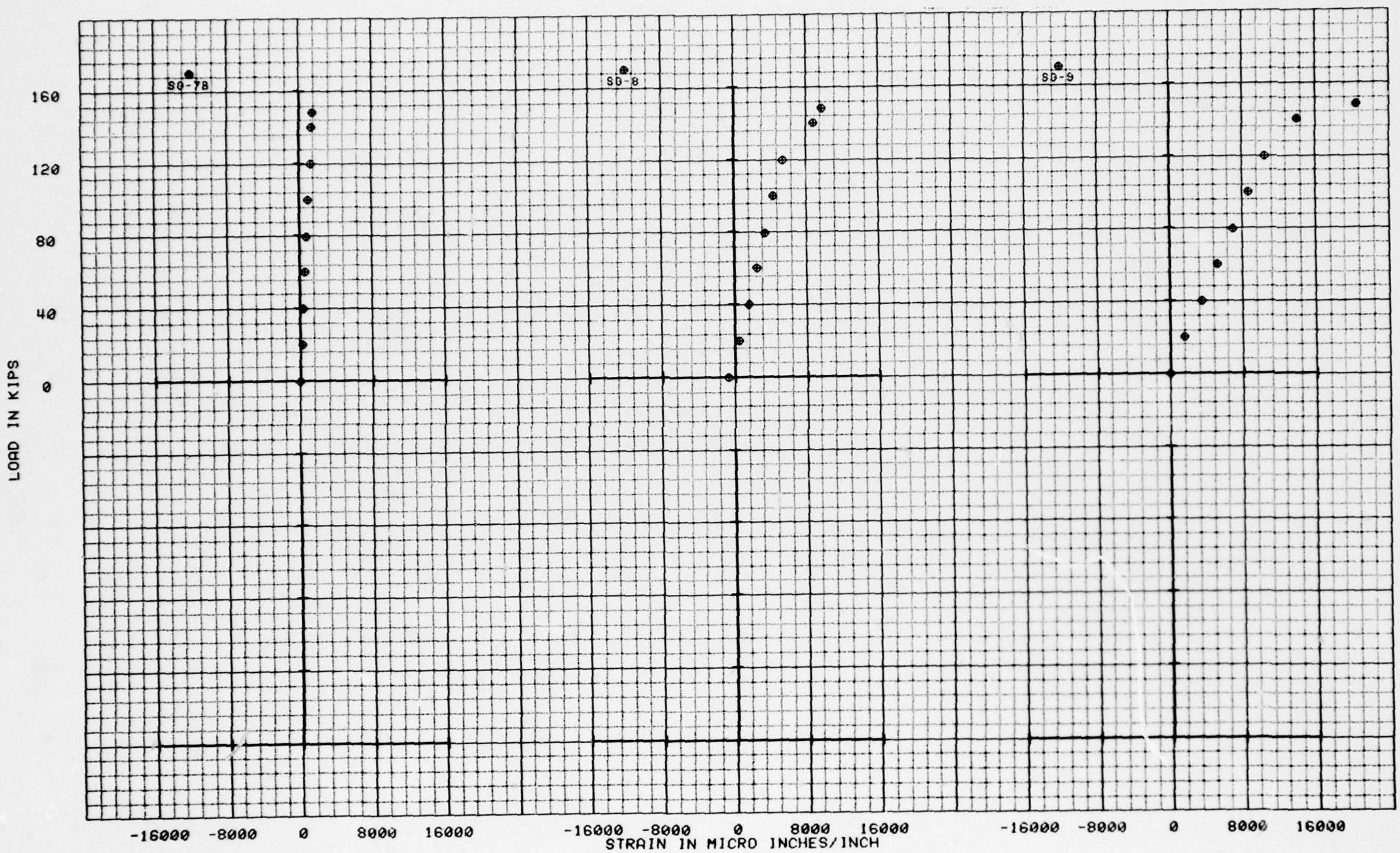


FIGURE B-27 Continued Strain Gage Data 75T060106-1013
1/2 Laminate Off Axis

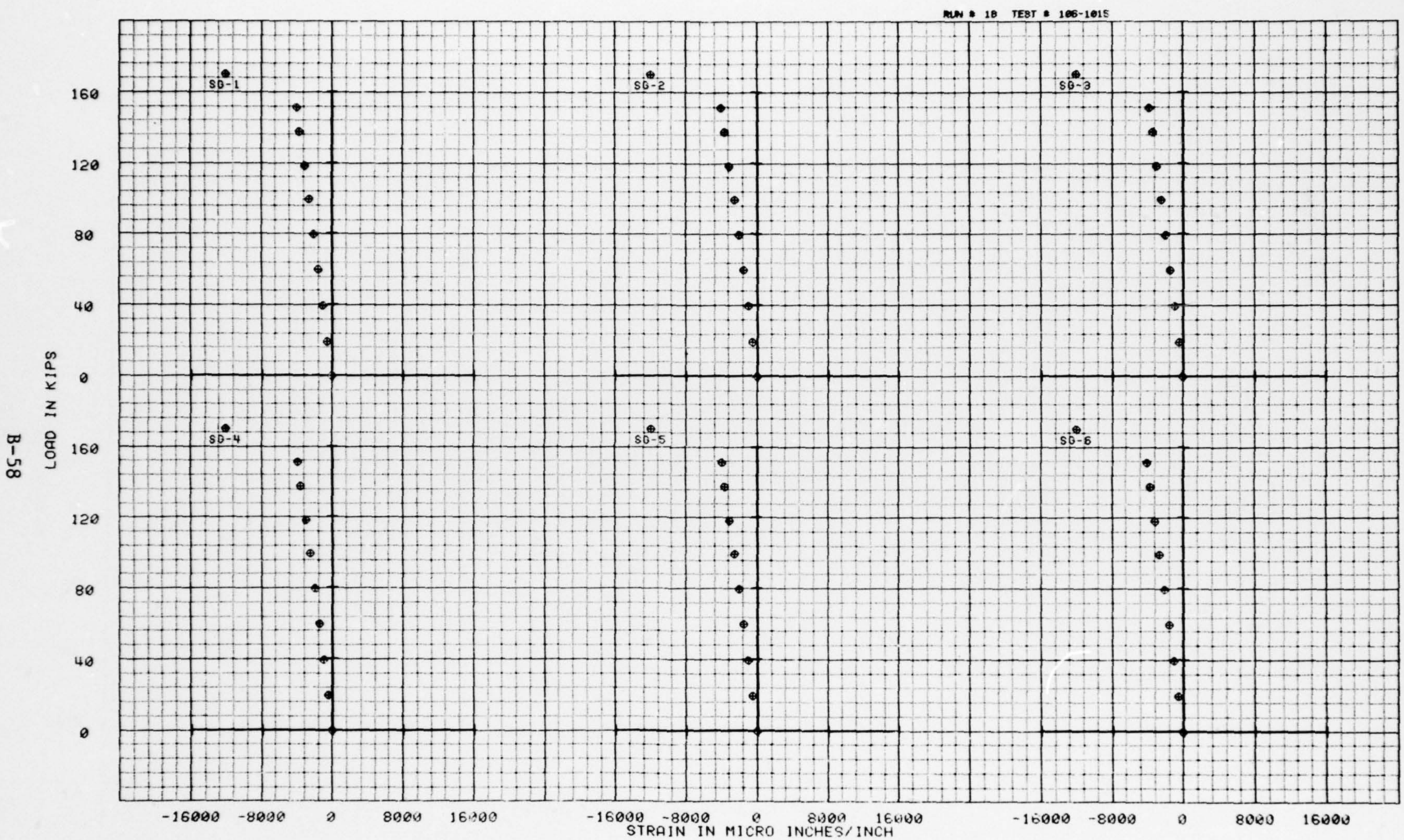


FIGURE B-28 Strain Gage Data 75T060106-1015
1/2 Laminate Compression

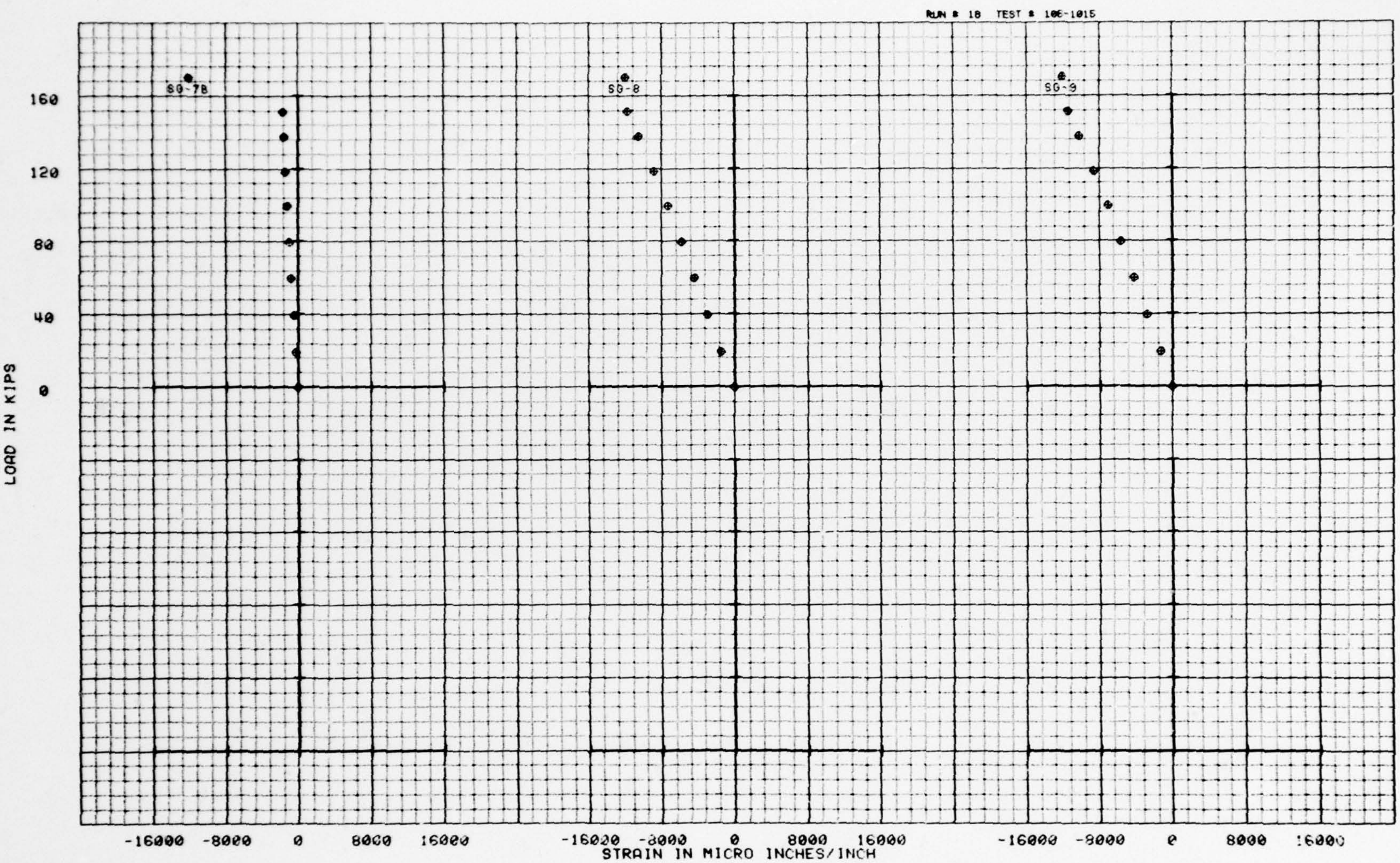


FIGURE B-28 Continued Strain Gage Data 75T060106-1015
1/2 Laminate Compression

APPENDIX C

THE COMPLIANCE METHOD

APPENDIX C

THE COMPLIANCE METHOD

In this approach, the skin and the repair patch are idealized as bar members interconnected by flexible fasteners. The in-plane shear lag in both the skin and patch is neglected, and it is assumed that the bolt loads produce a state of uniform deformation across the width of the members.

Figure C-1 schematically illustrates the approach for a repair concept with two rows of fasteners on each side of the hole in the skin. The rows are separated by an axial distance of a_1 and a_2. Each of the fasteners in Row 1 is loaded in shear by a bearing load P_{br_1}; likewise, each of the fasteners in Row 2 is acted upon by a bearing load P_{br_2}.

In the skin, the holes in Row 1 and Row 2 undergo an axial displacement of:

$$\delta_{1s} = \frac{a_1}{C_{1s} \; A_s \; E_s} \left[P - n_1 \, P_{br_1} - n_2 \, P_{br_2} \right] \qquad \text{(A-1)}$$

$$\delta_{2s} = \frac{a_2}{C_{2s} \; A_s \; E_s} \left[P - n_2 \, P_{br_2} \right] - \frac{a_1}{C_{1s} \; A_s \; E_s} \, n_1 \, P_{br_1} \qquad \text{(A-2)}$$

P = Applied Load in Skin

A_s = Cross-Sectional Area of Skin

E_s = Elastic Modulus of Skin

n_1 = Number of Fasteners in Row 1

n_2 = Number of Fasteners in Row 2

and C_{1s} and C_{2s} are the ratios at Rows 1 and 2, respectively of the displacement of an axially loaded strip with a hole of diameter D and the displacement of an unperforated strip. C_{1s} and C_{2s} account for the additional flexibility of the skin due to the hole and are given by:

$$C_{1s} = \frac{1}{1+\frac{3\pi D^2}{4a_1 W\phi}} \tag{A-3}$$

$$C_{2s} = \frac{1}{1+\frac{3\pi D^2}{4\ a_2\ W\phi}} \tag{A-4}$$

where:

$$\phi = 1-\frac{1}{2}\left(\frac{D}{W}\right)^2-\frac{1}{2}\left(\frac{D}{W}\right)^4 \tag{A-5}$$

and

D = Diameter of hole in skin
W = Width of skin

Similarly, the axial displacement of Rows 1 and 2 due to the flexibility of the patch and fasteners is:

$$\delta_{1_p} = \frac{a_1\ (n_1\ P_{br_1}+n_2\ P_{br_2})}{A_p\ E_p} + \frac{2P_{br_1}}{K_{bolt}} + \delta_o \tag{A-6}$$

and

$$\delta_{2_p} = \frac{a_2\ n_2\ P_{br_2}}{A_p\ E_p} + \frac{a_1\ n_1\ P_{br_2}}{A_p\ E_p} + \frac{2\ P_{br_2}}{k_{bolt}} + \delta_o \tag{A-7}$$

where:

A_p = Cross-sectional Area of Patch
E_p = Modulus of Patch
k_{bolt} = Bolt Stiffness
δ_o = Initial Fastener Clearance

For repair concepts where a backup plate is used for anchoring the nuts, the cross-sectional area of the plate is simply added to that for the repair patch.

In order for the displacements of the skin and the patch to be compatible, the following condition must be satisfied:

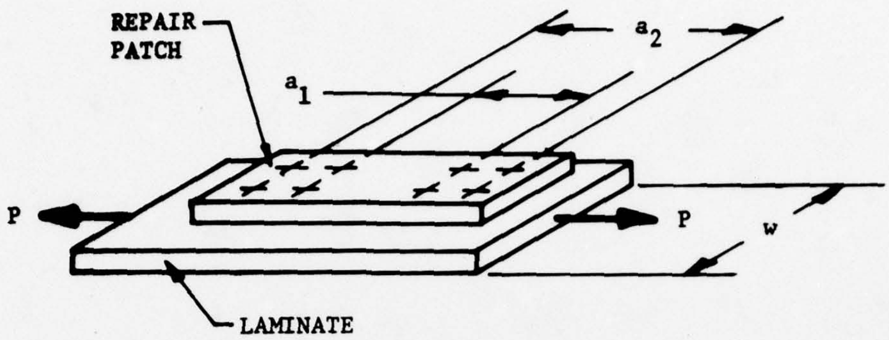
$$\delta_{1s} = \delta_{1p}$$
$$\delta_{2s} = \delta_{2p}$$

This leads to two simultaneous equations which must be solved for the unknown bearing loads; P_{br1} and P_{br2}. Once P_{br1} and P_{br2} are known, the maximum strain at the edge of the cutout is found from the equation:

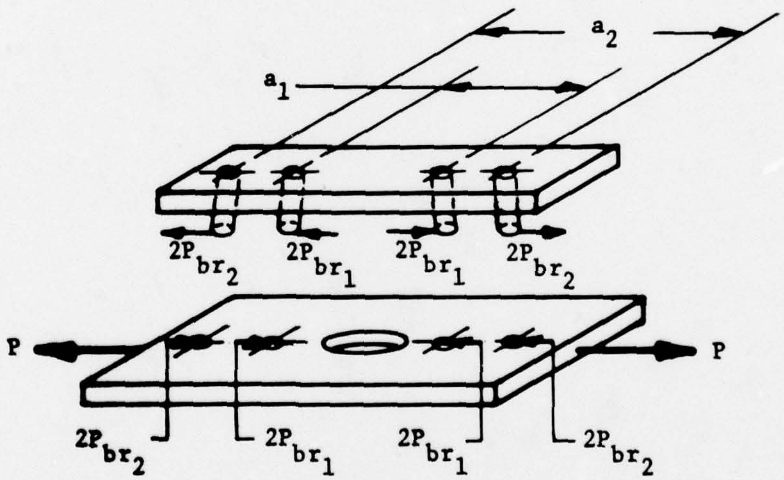
$$\varepsilon_{max} = \frac{(P - n_1 P_{br_1} - n_2 P_{br_2})}{A_s E_s} k_t \qquad \text{(A-8)}$$

where: K_t is the stress concentration factor for the hole. K_t is obtained by combining the theoretical value for a hole in an infinite orthotropic laminate (Ref 2) with the free edge correction factor of Reference (3).

Although the example mathematical derivation was for a repair concept involving two rows of fasteners on each side of the hole, the method can be easily extended to handle any numbers of rows.



REPAIRED SPECIMEN



COMPLIANCE MODEL IDEALIZATION

FIGURE C-1 COMPLIANCE METHOD REDUCES ANALYSIS OF REPAIR TO ONE DIMENSION